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TECHNICAL MEMORANDUM

X-650

THERMAL PROTECTION SYSTEMS

Report on the Aspects of Thermal Protection of Interest to

NASA and the Related Materials R & D Requirements

Prepared by the Panel on Thermal Protection of the
NASA Research Advisory Committee on Materials
October 1, 1961

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WASHINGTON

February 1962

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PANEL ON THERMAL PROTECTION

of the

NASA Research Advisory Committee on Materials

W. H. Steurer, Chairman
General Dynamics/Convair
San Diego, California

R. M. Crane
Ames Research Center
Moffett Field, California

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L. L. Gilbert
Aerojet-General Corporation
Azusa, California

C. A. Hermach
Ames Research Center
Moffett Field, California

E. Scala
AVCO Manufacturing Corporation
Wilmington, Massachusetts

E. J. Zeilberger
Rocketdyne Division
North American Aviation, Inc.
Canoga Park, California

Richard H. Raring, Secretary
NASA Headquarters
Washington, D. C.

[redacted]

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PREFACE

Recognizing the importance of Thermal Protection to NASA vehicle programs, as well as to aerospace technology in general, the NASA Research Advisory Committee on Materials established an ad hoc panel "to consider the various methods that have been proposed for heat alleviation of aerospace and power plant structures, with the objective of providing guidance for materials research."

The present report conveys a summary of the panel investigations and the resulting conclusions. It further includes the comments of a number of experts, who reviewed the first report draft.

The report does not attempt to give a complete account of the entire area of thermal protection. It is focused solely at the following two objectives:

(a) To give a general picture of the present state-of-the-art.

(b) To define the most important problem areas which are of significance for the various programs of NASA.

This limitation required considerable confinement in the treatment of the technical subjects involved. Prime emphasis was placed on the aspects of materials and material systems. Related topics such as aerodynamic considerations, specific trajectories and even the immediate aerothermal environment are only discussed to the extent to which they are needed to understand and assess the performance of materials. By the same token, design considerations as well as specific applications and shapes have only been included in a generalized form, wherever needed. It may further be noted that several significant areas of application such as ballistic nose cones are omitted, as they are not primarily the concern of NASA.

It is further recognized that most thermal protection systems are exposed to secondary environmental influences which may affect their thermal behavior. These have been disregarded intentionally with few exceptions, as their interaction can only be viewed in the light of specific cases. Besides, their discussion goes considerably beyond the scope of this report.

The numerical values used throughout the report should not be considered as design data, as they are merely typical numbers, introduced for the purpose of examples and the establishment of general relationships.

As the various sections have been prepared by various individuals, the report represents a variety of concepts, approaches and opinions.

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No attempt has been made to moderate seeming discrepancies between some chapters, as they are only part of the reasoning and do not affect the prime objective of the report, namely the definition of problem areas and the formulation of the pertinent R & D. Besides, this inconsistency is representative of the general disagreement between experts in this field, primarily resulting from the many disciplines and, therefore, schools of thought involved. It was further felt by the Panel, that this variety conveys the reader a true picture of the present situation and, at the same time, gives him the background information to form his own opinion.

Another inconsistency may be found in the symbols and dimensions used in various chapters, even though the necessity of uniform and standardized terms, symbols and dimensions is emphasized in the recommendations. It is however recognized that another screening of the terminology with the participation and the endorsement of all leading agencies will be required before a generally acceptable standard can be reached. In the meantime, modified symbols have to be retained in several areas of thermal protection in order to remain compatible with the language used in the connected disciplines.

The reader may, further, miss specific references to the statements made throughout the report. As the report is not a technical paper, the detailed reference work has been omitted. It was felt that this omission does not interfere with the purpose of the report.

It is finally emphasized that the report is primarily directed to the materials R & D community. Even though Thermal Protection involves many other disciplines, such as aero-thermodynamics, physics and chemistry, the problems are primarily judged from the viewpoint of materials and expressed in the language of the materials expert.

ACKNOWLEDGEMENTS

The Panel acknowledges the assistance obtained from members of the Advisory Committee on Materials in reviewing the report drafts. It acknowledges further a contribution prepared by staff members of the Missile and Space Division, Lockheed Aircraft Corporation, through Committee Member J. C. McDonald.

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TECHNICAL MEMORANDUM X-650

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GENERAL CONCLUSIONS

The present study has indicated clearly that the wide spectrum of requirements of future aerospace missions and vehicle designs calls for the development of a variety of thermal protection systems. Even though the general principles of each type of protection system are well advanced, it is also apparent that considerable more information and knowledge needs to be generated before optimum thermal protection systems can be reliably designed for many of the more critical applications. The following general conclusions appear to be in order.

1. Principles of Thermal Protection are well established.

2. Meaningful engineering properties and parameters for Thermal Protection Systems are not available.

3. Standardized test methods for thermal protection materials and the related test devices are non-existent.

4. Programmed environmental simulation tests will always be necessary to evaluate Thermal Protection Systems for specific flight requirements.

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*Title, Unclassified.

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PART I - STATUS REPORT

1. PRINCIPLES OF THERMAL PROTECTION

If a structure is exposed to heat, there are two ways to compensate for the decrease of physical properties in materials associated with elevated temperatures:

a. Selection of a material which retains sufficiently high physical properties in the specified temperature range,

b. Protection of the stress-carrying structure against the heat, so that it can be designed on the basis of conventional material properties. L650

The first way of dealing with high temperatures by means of so-called heat resistant materials is usually designated as the "<u>hot structure</u>." It is obvious that the hot structure has certain limits, as at very high temperatures practically all materials begin to lose their strength or their solidity. In addition to this, considerably lower temperature limitations than the ones indicated by the heat resistance of the materials are often dictated by functional requirements or payloads, such as instrumentation or a human crew. In both cases thermal protection becomes imperative.

The device by which this protection is achieved is called "Thermal Protection System" (in the following abbreviated by TPS). The TPS separates the heat resisting and stress-carrying function of a structure into an essentially non-load-carrying protection system, designed primarily for thermal capability, and a load-carrying substructure.[x] In its ideal form, the thermal protection system precludes any temperature rise in the stress-carrying substructure. In most practical cases however, the TPS acts merely as an impedance to the heat transfer, and a certain amount of heat is still absorbed by the substructure. The TPS will always be designed so that the peak temperatures in the substructure do not surpass permissible limits, indicated by the high temperature characteristics of materials or by payload requirements.

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[x]An ablative material may function as a load-bearing member in addition to its thermal protection function. This has been demonstrated experimentally in a USAF sponsored program with the Cornell Aeronautical Laboratory (74). (Numbers in parentheses refer to Reference List at the end of this report.)

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2. THERMAL ENVIRONMENT

a. Heat Sources

The thermal environment responsible for the heat input may be produced by external or internal heating sources. The external thermal environment refers primarily to the aerodynamic heating of a vehicle hull connected with high speeds of flight in the atmosphere as well as the solar heating in space. In addition, there may be secondary external sources such as the heating of the structure in the neighborhood of the
X650 tail flame of a propulsion system or the effect of nuclear explosions. Internal heating is primarily connected with propulsion or auxiliary power systems, as well as with heat released by electronic instrumentation.

b. Aerodynamic Heating

The amount of heat produced by aerodynamic heating depends primarily on the velocity and altitude of flight and, secondarily, on such parameters as the angle of attack and the flow pattern. In order to make comparisons, the variables may be fixed as in figure 1 where the convective heat transfer at the stagnation point for a one foot radius is shown for the entire altitude and velocity field. This chart is commonly used for the representation of the thermal environment for a specific vehicle type, by superimposing the trajectory to the thermal profile lines as a graphical background.

The heat flux or temperature lines represent steady state conditions, where a complete equilibrium between the heat input and the heat dissipated at the surface exists. As this state is attained only in long time operation, the chart is primarily used for the representation of the environment of cruise vehicles. The frequently practiced use for transient cases of heating, as encountered in re-entry vehicles, has to be taken with caution, as the chart neglects the time parameter, which is of prime importance for TPS.

c. Transient Aerodynamic Heating

As the mission times of vehicles decrease with increasing speeds, equilibrium conditions may no longer be obtained. In such cases, the heat flux values no longer apply accurately, as the heat transfer depends on the differential between the temperature in the boundary layer and the surface temperature. This transient heating is even more pronounced in all types of re-entry vehicles. In lift bodies, such as Dynasoar, the re-entry time may be in terms of fractions of an hour, while a drag body may encounter a heating of only minutes or seconds duration. In these cases certain arbitrary definitions have to be made for heat flux and time in order to obtain comparable values. After extensive considerations the formal Panel on Thermal Protection Systems of the MAB selected the peak heat flux $\dot{q}_{max}$ as representative for a non-uniform characteristic. The equivalent time is designated as $\bar{t}$ and obtained from the relationship $Q = \bar{t} \cdot \dot{q}_{max}$. By means of these values the original sinusoidal heat flux is replaced by a rectangular characteristic of equal total heat input Q, as illustrated in figure 2.

In order to arrive at comparable values, another definition has to be made. In transient heating cycles the heat input changes continuously with the temperature differential between the maximum temperature in the boundary layer and the temperature of the surface, as illustrated in figure 3. While for specific cases the material behavior has to be determined under exact assessment of the heating characteristics, a simplification is necessary to permit the establishment of generally applicable materials data. It has for this purpose become customary to represent the heat input in the so-called "cold wall" condition, i.e., room temperature. The equilibrium surface temperatures, derived from such cold wall heat flux values are therefore not accurate, however adequate for the appraisal of materials.

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With the so-defined values $\dot{q}_{max}$ and $\bar{t}$, the thermal environment of any vehicle exposed to aerodynamic heating can be expressed in terms of time and heat flux. This permits a direct comparison in a heat flux-time diagram as shown in figure 4. In general, the heating time becomes shorter as the heat flux is increasing, with the exception of some advanced manned re-entry vehicles, where considerable heat fluxes have to be sustained over a prolonged period of time. From the viewpoint of thermal protection, this condition represents the most severe case as will be shown in Part I-5.

d. Solar Heating

As we move to extreme altitudes, aerodynamic heating becomes extremely small due to the decreasing density of the atmosphere and, finally, in deep space the sun remains as the only source of heat. While in aerodynamic heating the heat flux is primarily convective, solar heating is exclusively of radiant nature. The rate of solar heating in the neighborhood of the earth is constant for all practical purposes and amounts to 0.12 Btu/ft^2/sec. This value changes however for various orbiting conditions in view of the radiation and reflection of thermal radiation from the earth and with the passage through its shadow.

The change of the heat flux across the planetary space depends on the distance from the sun. The following mean solar heat fluxes are encountered in the vicinity of the closer planets:

Venus	0.24 $Btu/ft^2/sec$
Mercury	0.79 $Btu/ft^2/sec$
Mars	0.052 $Btu/ft^2/sec$
Jupiter	0.0044 $Btu/ft^2/sec$
(Earth	0.12 $Btu/ft^2/sec$)

The thermal environment of space is discussed in more detail in Part II-3.

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External heating from other than natural sources, such as atomic explosions or the radiant heating from the tail flame of a propulsion system, are not considered here any further as the heating characteristics depend too much on the individual vehicle conditions.

e. Internal Heating

The most severe case of internal heating in astronautics engineering is encountered in rocket nozzles. This refers particularly to the throat area of solid propellant motors where, contrary to liquid propellants, cooling is usually prohibitive. Rocket nozzles and casings are discussed in detail in Part III-4.

The heating of the internal structure produced by propulsion systems, power units as well as by electronic instrumentation, is mostly conductive. Even though the heating rates are usually not very high, considerable thermal protection particularly in manned vehicles is required, as the dissipation of heat is somewhat more difficult than in external heating confined to the outer surface. Even though the thermal characteristics depend entirely on individual designs, certain selected cases are discussed in more detail in Part II-5 in view of their interest in NASA.

3. THERMAL PROTECTION SYSTEMS

It has become customary in engineering, to connect TPS's almost exclusively with aerodynamic heating. It is, however, apparent, that TPS's have a much broader spectrum of applications. They are therefore, appraised in this discussion from a very fundamental point of view.

a. General Criteria of Application

As the term implies, a TPS is always aimed at the protection of an object against heat. This object may be the structure itself, which requires protection in order to retain sufficient mechanical and other properties, or functional components, such as instruments or payloads, which may include a human crew. The selection, arrangement and dimensioning of a TPS is determined by the heat input from the environment

and the temperature which can be sustained by the component to be protected. If the TPS is solely aimed at temperature reduction in stress-carrying structures, it is obviously only applied externally, i.e., at the site of the heat input. In other cases where TPS serves the protection of functional components or humans, it may be arranged anywhere either externally or internally or even close to the component to be protected. Very often, however, the TPS has the combined task of protecting the structure as well as the payload; in such cases both external and internal thermal protection may be applied simultaneously. The proper apportioning of the heat reduction between external and internal protection systems depends on weight considerations, in which the resulting temperature of the intermediate stress-carrying structure plays X
a major role. In such combined systems entirely different schemes of 6
thermal protection may be used for the external and internal portion. 5
In the following, these partial TPS's are treated separately. 0

b. Classification

In carrying out the task of thermal protection, it is obvious that the law of conservation of energy has to be observed. Heat can only be either dissipated, i.e., rejected or removed from the place of heat input, or absorbed directly as heat, or by means of a process of energy conversion. On the basis of this fundamental concept of thermal protection, the known systems may be classified as follows:

TPS based on the principle of heat dissipation

- Direct radiation
- Insulation
- Indirect radiation
- Convective cooling

TPS based on the principle of heat absorption

- Heat sink
- Ablation
- Transpiration
- Film cooling
- Direct energy conversion

Heat Dissipation systems may be further distinguished with regard to the place of dissipation, which may be accomplished either by "direct dissipation" at the place of heat input (direct radiation, insulation), or by "indirect dissipation" at a place of lower heat input to which a portion of the heat is transferred either by conduction (indirect radiation) or by functional means (convective cooling).

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<u>Heat Absorption</u> may be "intrinsic" or "extrinsic," i.e., provided by the protective material itself (heat sink, ablation), or by a separately supplied coolant (film cooling, transpiration).

c. Systems Based on the Principle of Heat Dissipation

The primary tool of heat dissipation is radiation. This is preferably carried out at the surface of heat input. A certain portion of the heat will diffuse to the supporting structure and be absorbed by its thermal capacity. This process is, however, time limited: as the thermal capacity of these parts is exhausted, the heat conducted away from the surface of heat input is depending on the radiant heat dissipation at internal surfaces. Dissipation systems are primarily used for long time applications and equilibrium conditions are predominant, in which heat reduction is solely provided by radiation.

(1) Direct Radiation

The most elementary means of thermal protection is an increase of the surface emissivity by treatments or coatings. As in most cases heat transfer to other structural parts or a vehicle interior is undesirable, it is attempted to re-radiate to the environment as much heat as possible at the place of heat input. This may be enhanced by the use of internal insulation.

<u>Application</u>: It is apparent that for <u>aerodynamic heating</u> the thermal protection effect of direct radiation is limited by the deterioration of mechanical properties of materials at high temperature and by the requirements of thermal insulation of the interior from the hot radiating surface. For a moderate heat flux ($<\approx 25 \times$ Btu/ft^2/sec) and for relatively short-time applications, however, radiation is the most satisfactory and efficient method of thermal protection.

For <u>internal heating</u> highly reflective surfaces and coatings are extensively used to protect surrounding structures from a heat source.

For the <u>solar heating</u> of space vehicles, direct radiation is the primary means of thermal protection. As the heat source is purely radiative, the heat input into the surface can be reduced by low surface absorptivity. It can be further reduced by a high emissivity in the infrared, as space environment has no thermal capacity and heat can only be dissipated to the environment by radiation. The efficiency of a surface for thermal protection in space is commonly defined by the ratio α/ϵ of absorptivity for solar radiation to the emissivity for the longer wave length radiation emitted by the body. The α/ϵ ratios for a number of typical materials and surface conditions are listed in table 1.

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TABLE 1.- EMISSIVITIES OF SURFACES

Material	α	ϵ	α/ϵ
Stainless Steel 302			
Polished	0.50	0.15	3.33
Sandblasted	.71	.35	2.03
After 3d sandblast	.615	.47	1.46
2S Aluminum			
Polished	.31	.05	6
Sandblasted	.48	.33	1.45
Magnesium Alloy HK31			
Polished	.17	.11	1.65
Sandblasted	.63	.54	1.17
Rokide A .010" on SS 347	.21	.80	.26
Rokide A .020" on SS 347	.15	.77	.195
Solar cells covered by fused silics	.8	.84	.95
Solar cells (Hoffman)	.9	.39	2.30
Fiberglas-plastic (91-LD)	.2	.88	.23
Paint (Ti dioxide base)	.15	.90	.167
Tabor surface (i.e., Ni with thin layer of Ni oxide	.9	.1	9

(2) Insulation

Insulation is commonly considered as a means of heat reduction by the use of a low conductivity material or material system between the place of heat input and the structure or components to be protected. From the viewpoint of thermal protection systems, however, insulation is only a tool to concentrate the heat at the surface of the heat input and to increase the surface temperature and consequently the efficiency of radiant dissipation. The significant material properties of insulation systems are, therefore, the thermal conductivity of the bulk material and the temperature limitation of the surface with regard to melting or change of emissivity by oxidation.

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This combination of properties is primarily found in ceramic materials and pyrolytic graphite which exhibit a low thermal conductivity and a high temperature capability, even though the surface emissivities are usually low. In other insulation systems, non-metallic materials of low thermal conductivity in solid, particle or fiber form are contained in a metallic envelope which protects the insulating material mechanically. These systems are obviously limited by the temperature capability of the envelope. Various attempts are being made at present to produce materials of unusually low thermal conductivity which are however not yet perfected. In one system (Johns Manville Min-K) this is attempted by particle dimensions which provide internal gaps smaller than the mean free path of air so that conductivity heat transferred between these particles is minimized. Most of these advanced insulation materials are mechanically sensitive, particularly in the presence of vibrations.

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In table 2 the insulative parameter, ρK as a commonly accepted means of representing the insulating capacity, is listed for a number of materials and insulators. Insulation systems are not limited to the use of low conductivity materials. High insulation efficiency can be obtained in metallic systems if the configuration emphasizes heat rejection and maintains the conductive heat transfer at a low level. Basically, the entire heat in the surface panel of a multilayer system is turned into radiant heat, and the portion radiated at the inside surface redirected by means of a number of highly reflective layers (8). While such systems are comparatively insensitive to vibrations, their efficiency is limited due to the inherent partial conduction and the temperature limitations of the metallic components.

Even recently developed "advanced" insulation systems represent only marginal improvements. A major breakthrough with entirely new concepts is required to provide the urgently needed lightweight insulation for advanced high speed flight and re-entry vehicles.

Application: Internal insulation is, and has been, used extensively in connection with internal heat sources, as well as with external heating, including aerodynamic heating in high speed vehicles. The use of external insulation for high speed aircraft and re-entry vehicles is still approached with caution due to (1) limited mechanical properties, particularly in the presence of lift forces and vibrations, (2) its susceptibility to shear stresses occurring rearward of the stagnation point, (3) its sensitivity to moisture and (4) the difficult application to large and intricate surfaces. In spite of all these difficulties, external insulation will play a predominant role in the development of aircraft at speeds exceeding Mach 3, as well as of all types of re-entry or aerospace vehicles.

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TABLE 2.- TYPICAL PROPERTIES OF INSULATION MATERIALS

Material	Density, ρ lbs/ft^3	Thermal conductivity, k Btu/in hr ft^2	at °F	Insulative parameter, ρK Btu/lb hr ft^4	Temperature limit, °F	Heat flux limit, Btu/ft^2/sec
Thorium oxide (Th O_2) (Solid)	600	19.8	1,800	990	4,850	324
Zirconium oxide (ZR O_2) (Solid)	363	14.4	1,800	435	4,500	247
Forsterite (Mg_2SiO_4) 31% Porosity	139	11.6	1,800	134	3,400	90
Firebrick A (Solid)	65	3.6	2,000	19.5	3,000	58
Firebrick B (Solid)	29	1.27	1,500	3.07	1,800	10
Foamsil	11	1.85	900	1.70	2,000	15
Eccofoam	22	.75	800	1.38	1,600	7
Alumina coating (98.6% Al_2O_3)	200	19	2,000	317	3,000	52
Zirconia coating (98% ZrO_2)	325	8	2,000	216	4,200	192
Pyroceram 9608 (Solid)	156	13.6	70	177	2,250	22
Pyroceram (Foam)	14.4	1.5		------	2,250	22
Aluminum phosphate bonded alumina	190	7.2	2,400	114	3,500	100
Zirconium phosphate bonded zirconia	242	6.0	2,400	121	4,000	161
Min-K 2000 (Bonded powder, fiber reinf.)	20	.27	1,000	.45	2,000	15
Aluminum silicate fibers	6	3.0	1,300	1.5	2,300	23
Silica fibers (Q Felt)	3	1.16	1,400	.29	2,000	15
Basalt volcanic rock fibers	5	2.70	1,300	1.13	1,450	5
Diatomaceous earth powder	17.2	.89	1,600	1.28	2,500	31

(3) Indirect Dissipation

In structures which exhibit a preferred place of heat input, it may be desirable to provide a lateral diffusion of the heat away from the "hot spot," and to dissipate it by radiation at places of lower heat input. This, in effect, represents a thermal protection system with directional thermal conductivity, i.e., high conductivity in lateral direction and low conductivity normal to the surface, as it is found in pyrolytic graphite and eventually other pyrolytically deposited substances.

Application: In general, indirect radiation is indicated in all structures with a varying temperature profile, such as leading edges of aircraft wings, or missile fins. The application appears particularly attractive in space structures, where the "shadow side" is highly effective for radiant heat dissipation.

(4) Convective Cooling

The principle of indirect dissipation may be applied to large heated surfaces, if the previously directed passive mode of lateral heat diffusion is replaced by an active or functional heat transfer system. This is accomplished by a convective cooling system in which a circulating liquid transfers the heat from the wall to a radiator located at a place of low heat input and dissipated there to the environment. While in such systems considerable amounts of heat can be dissipated, they lead to complicated designs and are usually limited by weight considerations.

Extensive studies have been carried out on convective cooling systems, proposing water, lithium and hydrogen as potential cooling, i.e., heat transfer media.

Application: Convective cooling is, and has been extensively used for internal heating in connection with power plants or rocket nozzles. Its application to externally heated structures is and will remain limited in view of the complexity and high weight penalties.

A very vital application of convective cooling is a space radiator. Even though not exactly pertaining to TPS, the principles of heat exchange and the related material requirements are the source and developments that serve both applications.

There are three types of convective cooling systems: (a) circulating systems, in which the heat is dissipated at the heat exchanger primarily by radiation, (b) two-cycle circulating systems, in which the heat exchanger transfers the heat from one coolant to another, which in turn absorbs the heat by its thermal capacity to include vaporization, and (c) non-circulating systems, in which the coolant is retained in or adjacent to the wall to be cooled.

d. Systems Based on the Principle of Heat Absorption

If the heat input surpasses a certain amount, it can no longer be contained solely by heat dissipation and means of heat absorption have to be introduced. It is obvious that absorptive systems are limited in their time of operation as the heat absorbing capability is continuously expended and thus limited by weight or volume considerations. Absorptive systems are therefore preferably used for cases of purely transient heating, as encountered in missile re-entry, or to transient heating cycles superimposed on steady state conditions, as they occur during certain maneuvers of high speed aircraft.

(1) Heat Sink

A considerable amount of heat can be absorbed, in addition to radiant heat dissipation, by the skin itself if sufficient wall thickness of a material of high thermal capacity and diffusivity is provided. In its simplest form a "shell structure" is used in which the wall serves as a heat sink and stress-carrying structure simultaneously and does not require a substructure. Even though this stress-carrying heat sink will reach considerable temperatures, strength considerations are immaterial in view of the heavy wall thickness required for thermal reasons. In most cases however the heat absorbing and stress-carrying function of the structure has to be separated into two individual elements by means of an insulation layer in order to avoid excess heat transfer into the interior of the vehicle. The early developments of copper heat sinks are today superseded by the exclusive use of Beryllium and Graphite.

As any heat sink system is obviously limited by the melting temperature of the material, it is attempted to increase its efficiency by the use of heterogeneous materials, such as combinations of particles of metals and metal oxides. Heterogeneous heat sink systems are intended to combine the benefits of a high surface temperature and thus heat dissipation, with a satisfactory thermal capacity. Systems of Be and BeO have been found to be promising (1).

Heat sink systems have recently gained renewed attention due to the successful developments of pyrolytically deposited materials, particularly pyrolytic graphite. Due to its directional properties, it has the capability to dissipate the heat from the hot spot laterally to areas of lower heat input. The low normal conductivity, further, increases the surface temperature and, consequently, the heat dissipation by radiation, enhanced by the high emissivity of graphite.

Application: The use of heat sink for long-range ballistic missile nose cones is not practical. For nose cones of tactical missiles, the heat sink will remain attractive in view of its simplicity and ruggedness. The use for leading edges and nose cones of lift-type re-entry vehicles may rather be termed as "hot structure," as equilibrium conditions will prevail.

The most significant application of heat sink principles is in solid propellant rocket nozzles. However, many uses are already marginal and higher combustion temperatures of high energy propellants call for new approaches.

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(2) Ablation

Description: Ablation as a means of thermal protection is based on the principle of intrinsic absorption. Unlike the heat sink, which is confined to the thermal capacity of a material in the solid state, ablation utilizes in addition the heat of fusion, the heat content in the liquid state and the heat of vaporization, including a potential superheating of the vapor phase. The wide range of temperatures may well include other endothermic processes, such as phase changes, chemical reactions or decomposition. The potential of high surface temperatures further permits increased radiant heat dissipation.

Some of this radiated heat may be absorbed by the environment and potentially fed back into the system of heat exchange at the surface.

If the material is vaporized, the resulting change of the immediate environment may further reduce the heat transfer to the surface ("blocking action"), thus contributing to the efficiency of thermal protection. This is most effective in aerodynamic heating, due to the particular characteristics of the boundary layer.

The elements of heat absorption in ablation are, thus, as follows:

Heat content in the solid and liquid state
Heat of fusion
Heat of vaporization
Radiation
"Blocking" action
Decomposition, chemical reactions
Phase changes in the solid state
Superheating of the vapor phase

Even though it may seem obvious, it is emphasized that ablation is a process of heat absorption, in which the functional behavior of a material is of prime importance. Even though the thermal properties play a major role in providing the basic capabilities of a material, utilization depends on the proper functioning of the ablation process. The functional characteristics of a material cannot be described in terms of properties, as they usually represent a complex interaction between material and environment.

A typical example is the process of removal in the dynamic form of ablation as it is encountered in aerodynamic heating. A high heat of vaporization is meaningless, if removal under the mechanical effects of the gas flow occurs in the liquid phase. The material has to be designed for proper retention at the surface until the full thermal capacity has been expended.

As the process of removal is simultaneously affected by the environmental conditions, particularly gas pressure and velocity, as well as the flow pattern across the entire body, materials have to be specifically designed for every case.

<u>Materials</u>: While ablation, like all absorptive systems, is time limited and thus applicable to transient heating only, even the comparatively short operation times require special means to maintain the process in an orderly fashion for any length of time. This is primarily achieved by a steep thermal gradient, which confines the process to a minute surface layer. Materials of low thermal conductivity are, thus, imperative. There is, however, only a limited number of homogeneous materials, which at the same time possess the low conductivity and high thermal capacity. Composite and heterogeneous materials play, therefore, a predominant role in the design of materials with a custom-made combination of properties, which insure an efficient, orderly and continuous process of ablation.

In the design of composite ablative materials, the arrangement of the component materials, particularly the orientation with regard to the surface, influences the thermal efficiency considerably. This has been amply demonstrated in early tests on re-inforced plastics with varied fiber orientation. The design of ablation materials has further to be compatible with the body geometry, which dictates the flow pattern and the heat input profile.

The temperature of operation and consequently, heat absorbing phenomena utilized in a specific ablating TPS, can be well governed by the design of the material. There are three basic operational conditions:

(a) The liquid system, in which the ablation process takes place at a temperature level between melting and boiling point

(b) The gaseous system at a temperature at or above the boiling point

(c) The semi-gaseous sytem, which is basically a liquid system, in which vaporization takes place while the liquid material is moving along the surface. In this case, the original place of ablation does not benefit from the vaporization.

As a rule, the operating temperature is attained rather fast, in view of the steep thermal gradient. From this point on, "quasi-steady state" conditions exist, as the process matches automatically any heat flux by means of the ablation rate, which with it proceeds normal to the surface, rather than to affect the surface conditions. This independence of the surface phenomena and the self-adjustment to any heat input, within reasonable limits, is one of the outstanding characteristics of ablation, as it automatically matches any form of heating cycle.

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Application: Derived from experiences in early applications in rocket jet vanes, high heat flux ablation systems were first perfected for long-range ballistic missile nose cones, where they are used today almost exclusively. For lower heat fluxes, as they are encountered in Missile Defense System, "slow ablation" systems have been developed. High speed aircraft and, particularly, re-entry vehicles have many potential uses for ablation systems, yet development is not in pace with requirements. For many of such vehicles, particularly those which exhibit long heating times at low heat flux levels combined with short peaks at high levels, "partial ablation" may be attractive, i.e., a material which acts primarily as an insulator, yet has ablation capability at higher heating rates. Increased use in solid propellant rocket nozzles is also expected.

(3) Transpiration

While in heat sink and ablation the thermal capacity is provided by the skin material itself (intrinsic absorption) and consequently limited, increased heat absorption and longer operation can be achieved by extrinsic absorption. The only fairly developed extrinsic system is transpiration cooling in which the heat absorbing effected by vaporization of a liquid is supplied through a porous wall at the surface of the heat input. The efficiency of transpiration systems with regard to heat flux and time is limited by weight considerations, even though transpiration systems appear to become superior to other absorptive systems at long operation times and extreme heat fluxes. It should be noted that in such comparison the weight of the transpiration system includes the necessary tankage, pressurizing and distribution hardware.

(Glass)

For single applications and short operation times, transpiration cooling may also be produced intrinsically. Such systems incorporate low vaporization point subliming materials having high heats of sublimation in the surface layers in such distribution as to retain structural strength despite loss of the subliming material.[x]

Application: So far, transpiration cooling has not been used except in liquid propellant rocket nozzles, and no immediate applications are foreseen. It may, however, be the only resort at extreme heat fluxes, where ablation is not acceptable, either because of long time requirements, or in view of the ablation products.

(4) Film Cooling

Heat absorption may be achieved also by means of a liquid or gaseous film supplied through a porous wall and maintained on the surface of heat input. It requires a moving environment, as it is present in aerodynamic applications, in order to maintain film flow and uniformity. X650

While liquid film cooling systems rely exclusively on the thermal capacity of the coolant, the gaseous system may have a considerable growth potential. By proper adaption of the gas injection system to the particular aerodynamic characteristics of the surface, the gaseous film may be very effective in reducing the heat transfer to the surface by thickening of the boundary layer. This is no longer an absorptive means of thermal protection and actually introduces "boundary layer modification" as a third principle of thermal protection.

Application: So far film cooling has been limited to experimental work. Boundary layer control appears to be very promising for lift-type re-entry applications, as outlined in Part II.

(5) Direct Energy Conversion

In space it would be desirable to utilize the solar heating of a vehicle hull for power generation rather than to dissipate it to the environment. This indicates the potential of a TPS, in which the heat is transformed into electrical energy and absorbed indirectly at the place of power utilization. No definite systems have been proposed yet. Reference is made to the report of the Ad Hoc Panel on Direct Energy Conversion of the NASA Committee on Materials.

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[x]This was experimentally shown feasible in the Mat'ls Cent. three years ago when subliming salts were impregnated in graphite and exposed to a fairly high heat flux. The rear surface temperature of a slab of graphite so impregnated remained substantially cooler than a non-impregnated slab for the same exposure time. Advantage: No liquid pumping, tanks or circulating system required.

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(6) Magneto-Hydrodynamics

It has been proposed repeatedly to use a magnetic field to repel the hot ionized gases as a means of thermal protection for high heating rates and prolonged time periods. This potential application of MHD for reducing convective heat transfer is currently being investigated by a number of organizations.

e. Capabilities of Thermal-Protection Systems

(1) Limitations

From the foregoing discussion of the various types of thermal protection systems, a certain number of fundamental characteristics as to their capability can be derived. While the heat flux which can be contained by dissipation systems is clearly limited, their operation time is almost unlimited as they operate on the basis of steady state conditions. Absorptive systems, on the other hand, are capable of absorbing considerable amounts of heat, yet are limited in operation time.

These limitations are primarily dictated by weight considerations, as weight is the prime design criterion in aerospace engineering. While in dissipation systems weight limitations determine the thermal efficiency, i.e., the highest heat flux which can be sustained, weight can be traded in absorptive systems against heat flux as well as time. The overall efficiency of a TPS is therefore determined by the interrelation of heat flux, operation time and weight requirements.

(2) Weight Considerations

Before a weight-related comparison of TPS can be made, it has to be defined, how much of the skin structure is included in the weight of the TPS. For this purpose the following definitions have been made (fig. 5):

W_t "Total weight" of both the TPS and the directly connected stress-carrying sub-structure

W_g "Gross weight" of the entire protection system, excluding the stress-carrying sub-structure

W_n "Net weight," representing only the portion of the TPS actively participating in the process of heat absorption or dissipation, e.g., the portion removed in ablation or the expended coolant in transpiration

 [redacted]

Δw "Support weight," comprising the part of the protection system, which supports its functioning indirectly, such as the thickness remaining at the end of the ablation process, or the coolant tanks, pumps, plumbing and injection systems in transpiration

As TPS refers always to a surface, weight is expressed per unit of surface area (lbs/ft^2).

The most representative weight expression is the gross weight W_g, as it identifies the weight penalties of thermal protection in vehicle X
design. For passive systems (insulations, heat sink) this presents no 6
transfer calculations. In all active systems (convective cooling, abla- 5
tion, etc.) the net weight W_n is either measured in tests or calculated 0
from previously measured material properties, such as Q^*. The value Δw, which complements W_n to attain the gross weight W_g, cannot be measured or accurately calculated and often has to be based on design experience. In the following table 3 some comparative values for Δw are listed together with an identification of the components of the TPS which comprise the support weight.

TABLE 3.- SUGGESTED SUPPORT WEIGHT REQUIREMENTS

Δw OF THERMAL PROTECTION SYSTEMS

System	Support components	Δw			
Convective cooling	Double wall Heat exchanger Plumbing, Controls Pumps including power supply	15 lbs/ft^2			
Ablation	Non-ablating wall thickness for insulation and reliability	(in % of ablating weight W_n)			
		Time sec.	20	40	over 120
		Nonmetallic composite	40	25	20
		Metal-ceramic graphite	65	45	35
		Ceramics	50	35	25
Transpiration	Porous wall Coolant tanks Plumbing, Controls Pumps including power supply	10 lbs/ft^2			
Film cooling	Porous wall Coolant tanks Plumbing, Controls Pumps, including power supply	10 lbs/ft^2			

[redacted]

DEC [redacted] IFIED

(3) Thermal Capabilities

If the capability of a thermal protection system is determined by either test or calculation, the immediately resulting weight requirements refer to the net weight W_n, as only the portion of the TPS is assessed, which is directly engaged in the process of heat transfer reduction. The relationship of heat flux $\dot{q}$, time t and weight W_n, which constitutes the thermal capability of a TPS, is preferably represented in a $\dot{q} - W_n$ diagram with time t as parameter. As most dissipation systems are almost insensitive to operation time, the thermal capability is sufficiently defined by a single line relationship. The time dependency of absorptive systems has to be taken in account by means of a set of lines for various time limitations.

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A typical $\dot{q} - W_n$ diagram is shown in figure 6. Most TPS's exhibit a straight line relationship over a wide range of heat fluxes. This linear portion which covers in most cases the useful range of the TPS, lends itself to the establishment of a convenient value representing the thermal capability of a TPS or a material. This property has been termed by Panel on TPS of the MAB as "Effective Thermal Capacity Q^*" in Btu/lb, identifying the amount of heat, which can be absorbed by a pound of a material or the thermally active part of a TPS. It is defined by the simple relationship:

$$Q^* = \frac{\dot{q} \cdot t}{W_n} \qquad \text{(Btu/lb)} \qquad (1)$$

With no further qualification, Q^* identifies for a single material the entire effective thermal capacity to include vaporization. For TPS it applies to the useful range only, which has to be specified by suffixes, as indicated in figure 6. It is emphasized, that this property is not identical to the heat content of a material, as it refers specifically to its use in a TPS and includes therefore, all secondary parameters and interactions with the environment. For this reason, its determination by calculation is limited to a few cases, where the process of heat reduction can be defined accurately. In most cases, the system of thermal, physical and chemical interactions is too complex for a reliable analysis, and Q^* has to be established in simulation tests.

Once Q^* is so established, it permits a fairly reliable prediction of the weight requirements for a TPS for other combinations of $\dot{q}$ and t, than the ones used in simulation tests, as long as the combination is within the range indicated by suffix. For ablation systems, e.g., the ablation rate can be derived from Q^* by the modified relation (1)

$$\dot{a} = \frac{\dot{q}}{\rho \cdot Q^*} \qquad \text{(in/sec)} \qquad (2a)$$

[redacted]

and the thickness expended for a given operation time by

$$a = \frac{\dot{q} \cdot t}{\rho \cdot Q^*} \quad \text{(in)} \qquad (2b)$$

The various methods for the determination of Q^* by either test or calculation or both are described in Part I-4.

(4) Design Efficiency

While Q^* and the $\dot{q} - W_n$ diagram identify the thermal capabilities of a material or TPS, its overall efficiency in design has to take all the support weight requirements in account. For this purpose, W_n has to be replaced by W_g by adding the value Δw. In some cases, as in ablation, Δw is related to W_n; in most cases it can, in the first approach, be assessed with a bulk figure, as listed in table 3. With introduction of the so obtained gross weight requirements W_g, the $\dot{q} - W_g$ diagram conveys the most complete picture of the relative efficiency of TPS's in weight-conscious design. X 650

A typical example of this representation is shown in figure 7 of a composite fiberglass-phenolic ablation system. The initial slope of the curves identifies the W_n-related support weight requirements Δw.

An overall comparison of various TPS's, based on available typical data, is presented in figure 8 for an arbitrarily selected time-limit of 40 seconds. Several dissipation systems are shown once more separately in a more appropriate $\dot{q}$-scale (fig. 9) for three time limits. Of particular significance in this comparison are the intersect points of the efficiency lines, as they identify the heat flux, at which, for a given time, one TPS becomes superior to another.

It is often beneficial to investigate the time influence more accurately by transforming the $\dot{q} - W_g$ diagram into a $t - W_g$ diagram with $\dot{q}$ as parameter. In the example of a comparison of transpiration and ablation systems (fig. 10), the significance of the intersect points is evident. For a heat flux $\dot{q}$ of 2,000 Btu/ft^2 sec, transpiration becomes superior to a SiO_2 ablation system at 16 seconds, yet is still heavier than a graphite ablation system, which on the other hand is not as perfected as the SiO_2 system. The second intersect point at 55 seconds, finally, marks the expected superiority of transpiration for long time operation.

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4. DESIGN INTEGRATION

a. General Considerations

The matching of an engineering material, or its performance, with the environment and operational requirements is defined as design integration. Several factors are involved, most of which have been discussed in previous critiques of thermal protection systems or the environmental analyses. In conventional engineering the selection of a particular material for a given design is based upon relatively simple environmental factors such as load, and materials compatibility is expressed in terms of design criteria. In thermal protection, however, the severity and complexity of thermal environments and their interaction with statistically proven materials criteria have created the requirement for a new approach. There is little or no precedent, in terms of design allowables, or even comparative performance data, which might provide a design engineer the essential basis for decision in applying a thermal protection system to an aircraft or missile structure.

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The available information is in the form of aerodynamic, thermodynamics, and chemical manipulations frequently unintelligible or unacceptable to other investigators in the field. The data are generally useless to design engineers and materials engineers. It is only within the past year that any generally used or accepted system of nomenclature has been observed. There is now an immediate need for simple precise methods for comparing and intercomparing thermal protection systems at several intensities of thermal environments. This will rapidly lead to understandable and useful design data for structural application.

If this is to be accomplished the present methods of assessing and evaluating thermal protection systems must be examined. From such a critical study the approaches to a sound materials test technique should be evident. This method and the data resulting from it should have a sound basis in present theory. However, it should not be entirely dependent on current technical uncertainties to the degree that minor qualifying assumptions are allowed to confuse or degrade the usefulness of the data. Present activity in the assessment of thermal protection systems may be arbitrarily separated into three areas. These are (1) a theoretical approach (2) an experimental approach and (3) a combination of these designated as the technological approach.

b. Theoretical Method

This approach to the analysis of thermal protection systems arises from the practical necessity of producing data in the absence of experimentally verified fact. The techniques were very successfully employed by the theoretical aerodynamicists in defining the hyperthermal environment and heating rates associated with hypersonic re-entry (54-61).

This was possible because of some precedent and the relatively low number of variables involved. Most important, it was the only way of determining any knowledge of the environment.

(1) Background

The theoretical approach to thermal protection system analysis is
an outgrowth of the aerodynamic model. The methods are reasonably valid
when applied to passive systems where the material does not actively
participate in boundary layer interaction, namely insulative-radiative
systems, heat sinks and convectively cooled systems. Indeed many of the X
early studies on ballistic re-entry and materials performance (heat sink) 6
were based upon heat transfer theories and the thermophysical properties 5
of the materials involved. The transfer of heat in matter is not a pro- 0
found topic and several excellent references are available which permit
rather accurate prediction of performance. Furthermore, these same methods
are exceptionally useful in the measurement of the environmental prop-
erties of experimental test equipment (62, 63). There are possible
sources of error in these purely analytical methods, particularly in
cases involving the reduction of the theory to experimental practice.

(2) Application

Since the demonstration of practical, engineering ablation systems for re-entry nose cones, there has been a flood of theoretical analyses attempting to provide macroscopic and microscopic justification for the ablation mechanism. Such studies may be divided into three distinct areas; (1) theoretical gas dynamics, aerodynamics, and thermodynamics (54-61), (2) chemical kinetic theory generally in the area of the ablation process, and (3) theoretical evaluations of tests, both good and bad (54-67), and the use of (1) and (2) above to justify the expression of test results (68-70). It is generally agreed that these are most interesting studies in chemical or gas kinetics and provide an occasional useful function in relating or understanding the reactions of a material to the environment. They may also be valuable in uncovering phenomena which, in the areas of transpiration or ablation, contribute to the increased effectiveness of the process. It should be borne in mind however, that this type of study is in the domain of the chemical thermodynamicist or the theoretical gas dynamicist. Such study is frequently more cultural than progressive when viewed from the more practical aspects of materials engineering. While support in these areas is by no means discouraged it should be comparable to the chemical kinetic research which has accompanied combustion engineering for the past fifty years.

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(3) Expression of Results

There still exists, both in materials and more sophisticated circles, a large degree of confusion regarding the expression of results nomenclature, units and correlation functions. It is not the purpose of this review to examine all the possible theoretical facets of ablation theory or even the errors involved in interpolating theory into practical engineering. The intent of this survey is to recommend an understandable system of units and a logical approach to test conditions for materials and design engineers. These must be reasonably simple and accurate. This study cannot evaluate all of the detailed considerations that are necessary for doing research into hypersonic heat transfer, gas dynamics or chemical kinetics. The present confusion rests to a large degree on a persistant attitude to interpret materials and design requirements in terms of a research philosophy. There is a middle ground of sound engineering application which involves judgement and the design of materials into systems. In this area there is a requirement for design data which will permit the selection of a proper material suitable for a particular environment. Nomenclature, units and tests are required to provide these data. While most engineers are adaptable to new environments and the necessary knowledge to understand them, this does not involve the conversion of all engineers into theoretical dynamicists or vice-versa. A basic engineering approach will be the goal of this review.

(4) Validity

In summary it must be emphasized that a theoretical approach to thermal protection is necessary and valuable. It is needed to direct research in the improvement of systems by establishing the fundamental mechanisms of operation. It assists the purely engineering approach to thermal protection and delineates the important parameters which can be generalized for an understandable approach to the generation of design data. The immediate danger of these theoretical systems is the tendency to over-complicate a clear cut and reasonably accurate engineering understanding.

c. Experimental Method

This method is a common approach to the solution of engineering problems. It is still widely used and as such was responsible for the first application of the ablation thermal protection system on a re-entry nose cone. It involves extensive testing, guided by intelligent engineering assumptions, but without the benefit of detailed theoretical analysis. This approach is an essential part of engineering and few practical designs are placed in production without such testing irrespective of the availability of theoretical analyses or even design data.

[redacted]

(1) Background

As early as 1949 rocket exhaust tests demonstrated that laminated plastics would outlast metals by a factor of 2-10. Working along similar lines materials personnel at the Army Ballistics Missile Agency at Huntsville, Alabama reasoned that matter on a change of state was capable of absorbing far greater quantities of energy than that absorbed in the solid state conditions. Systematic investigation of materials performance in rocket exhausts quickly demonstrated the comparative thermal protective capability of a broad range of materials. From this work came the first successful thermal protection engineering, the square wave heat pulse, the $\dot{q}$ (Btu/ft^2-sec) heat flux convention, Q^* (Btu/lb) thermal capacity, ablation rate data and much of the basic engineering knowledge which now has become the topic of detailed theoretical study. The experimental approach has many pitfalls but has the advantage of producing engineering data with the least possible delay. Furthermore, regardless of the initial approach, sooner or later the models must be tested if engineering is to be converted into hardware.

(2) Validity

The disadvantages to this wrench-in-hand approach are those which involve too hasty action with insufficient planning or study. In the case at hand, the accuracy of environmental simulation is a critical factor still partially unresolved. This involves the flow, chemistry, heat, and time of the environment or exposure. The results obtained in a free jet of combustible gas or even a plasmajet can produce substantially different results than those from a rocket exhaust nozzle even if the chemistry were to be ignored. The shock tube, because of a time limitation is totally inadequate for materials studies. On the other hand, a fortuitous combination of incorrect simulation and incorrect measurements may produce a correct result. As a typical example, it has been observed at Convair that stagnation point heating rates may be misleading by as much as 100% (63); this error may be automatically corrected by a calorimetric measurement, the effects of improper chemistry, a dissimilarity in shape or configuration, or by incorrect simulation of mass flow. Furthermore, if reasonable tests facilities are not immediately available the cost and time involved can be a prohibitive limitation.

d. Technological Approach (Recommended)

The status of both theory and experiment on thermal protection systems is still very inadequate for engineering application. This is probably due to the present over-emphasis of theoretical approaches, particularly in the area of ablation, to the exclusion of design data type development of all thermal protection systems. Based upon the information that is available a reasonably acceptable test procedure may be outlined. The results of these tests are checked at frequent intervals for confirmation with theory. This is the basis of the technological approach to engineered thermal protection systems.

(1) Major Test Requirements

The outstanding requirement for test data pertaining to thermal protection systems is that involving design criteria and comparative performance information for various competitive systems. Preferably, these data should be expressed in terms of weight required for a specific period of protection or level of protection.

These data further imply the need for accurate reproducible testing which can be duplicated in several laboratories. Several techniques are now in existence (63, 69, 71-74), many of which are useless, or of questionable value. In order to qualify a test, several factors must be considered. These include:

a. Environmental simulation
b. Test criteria
c. Sample design
d. Test measurements
e. Meaningful expression of results

As previously indicated, any test and the validity of the data should conform to the best accepted theoretical considerations. The accomplishment of tests which do not consider the above factors, or the interpretation of theory without careful test procedures has created much of the confusion which presently exists in thermal protection concepts. Each of the five criteria of the generation of test data will be examined and where possible correlation with existing theory will be cited. The goal of this correlation will be the synthesis of design criteria through an accepted test plan and finally a method of expressing and correlating data.

(2) Environmental Simulation

The aerodynamic types of thermal protection systems have four environmental factors which must be considered in any type of simulation. These are:

1. Heat
2. Flow
3. Chemistry
4. Time

The space vehicle will also have these environments if re-entry into the earth's atmosphere is involved, and occasionally during the launch phase. While in space a thermal regulatory protection system must also be considered. The primary environmental factors here are exceedingly low partial pressures of gas and a spectral and intensity distribution of electromagnetic energy which is still experimentally impossible. High energy atomic particles or meteoric type particles may also have a secondary influence on the thermal protection performance. Only the primary factors of aerodynamic heating will be considered in this discussion.

Heat

There have been several excellent theoretical aerodynamic studies and shock tube experiments which have, to a substantial degree, defined the environmental intensity of incident aerodynamic heating (54-61). Unfortunately, the shock tube does not qualify as a material test device. The commonly accepted unit of heat intensity is the Btu/ft^2sec now designated as $\dot{q}$.

Aerodynamic heat is transferred by two mechanisms, convection and radiation. Simulation by radiant heating alone is not adequate for several thermal protection mechanisms, although it is still limitedly useful in structural tests of relatively large size and low heat transfer rates. The other aspect of radiant heating involves the simulation of solar electromagnetic radiation. In these cases the absolute intensities, intensity distribution and spectral distribution are of prime importance (75). Spectral distribution may also be very important where very high aerodynamic heat fluxes are involved.

In any type of consumable thermal protection mechanism convective heat transfer is a vital part of the process and no simulation is adequate unless this type of environment is present. There are cases involving radiative protective processes and problems associated with oxidation rates where convective heat transfer is necessary.

Finally, the heating rate which is implied in the heat flux is frequently confused or misinterpreted, particularly in radiant tests of large mass structural applications. These heat flux requirements should also be accompanied by temperature rise data.

There are now available elaborate analyses, both dimensional and non-dimensional regarding the behavior of ablation materials under the influence of a high enthalpy-mass flow, or high flux environment (64, 68). Another program (76) (Plasmadyne) is also underway to experimentally establish the influence of various parameters on heat transfer and materials performance. Data are both conflicting and complex. At present it is customary to use the heat flux or heat transfer rate to a cold (water cooled) calorimeter as a standard for the heat environment. This is the so-called $\dot{q}_{max}$. It is related to heat content or enthalpy by the following relation

$$\dot{q} = \int (h \cdot \dot{m}) \tag{2}$$

where

$\dot{q}$ = heat flux (Btu/ft^{2}sec)

h = environmental gas enthalpy (Btu/lb)

$\dot{m}$ = mass flow rate (lb/ft^{2}sec)

In spite of its convenience and wide application, this representation of the thermal environment is by no means accurate. It has to be considered, for instance, that a blunt body behaves like a hemisphere-cylinder so that the stagnation point value cannot be applied to an arbitrary model. Further, for arcs, equilibrium flow may prove to be the best means for calibration of the stream. However, at the present state-of-art, no method can be considered as a general standard.

Attempts have been made to relate materials performance with both heat flux $\dot{q}$ and stagnation enthalpy h_s. A broad area of inaccuracy and confusion surrounds both methods. Again much of the argument enters around ablation theory rather than a more general comparison of materials and techniques for refining the environment conditions and/or measurement. The one greatest fallacy that exists in present test techniques involving high heat inputs, is the use of and the interpretation of data produced in free jets with and without graphite contamination.* A close second is the reliance on calorimetric measurements which have not been carefully checked for accuracy and application to the experimental conditions. The amount of heat in the environment, i.e., enthalpy is the most critical factor in the entire study. It should be carefully defined, understood and measured if any consistent system of material evaluation is to come from these investigations.

<u>Flow</u>

Except for the singular case of thermal protection in space all systems are concerned with heat transfer across boundary layers set up

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*The effect of air arc plasma (contaminated with electrode carbon) on oxidizable ablative materials has been investigated. A good report on this subject is AFBMD-TN-60-3 "On Carbon Contamination of Air Arcs and Its Effects on Ablation Measurements." With continued development on non-erodible electrodes, the [redacted] contamination may be eliminated.

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in supersonic or hypersonic flow. There are several factors to be considered where flow is involved. These factors are the effects of convective heat transfer or its interruption, the total and dynamic pressures involved, oxidation rates of materials in static versus dynamic environments, as well as questions involving subsonic versus supersonic flow.

It should be first recognized that practically all hypersonic heat simulation testing is accomplished by accepting a comparatively high enthalpy, low mass-flow combination. In actual flight environments, higher mass flows at lower enthalpy are usually encountered. General theory and some few isolated experiments tend to confirm the validity of this similarity assumption. The real reason for doing this is the unavailability of hypersonic nozzle materials which can produce this type of sustained flow and withstand the enormous heat transfer rates involved. While this limitation probably does not affect heat transfer in any major way it may very well influence the static or total pressure. The effect of pressure on such process as ablation has been treated theoretically (64). Few if any data exist from experimental studies. It must be reasoned however that in cases where chemical reaction rates are involved, both temperature and pressure influence the rate constants. This may not be appreciable however when reactions take place in the solid or liquid phases. Perhaps a more significant effect of pressure is its effect on gas evolution and diffusion which tends to decrease the so-called "blocking action," producing thinner boundary layers and increasing heat transfer. There would therefore, be a tendency for one condition to counter balance the other.

A closely related problem is the present uncertainty regarding the requirements for subsonic versus supersonic flow. As in the case of pressure this argument may start from the first law of thermodynamics where

$$E = q + pv$$

Since at the stagnation point the pv product degenerates to heat there should be little difference between a high subsonic and a low supersonic heat flow. As previously stated the major difference may be that associated with high velocity or the resultant pressures which could influence the heat transfer across boundary layers associated with gas diffusion and the removal of insulating, emissive or otherwise protective char layers (52).

In instances where reactions with oxygen or nitrogen at higher temperatures are involved, the requirement for dynamic flow conditions is mandatory. The chemical reaction of a metal or carbon at high temperature is relatively rapid. The displacement of the reaction by reaction product removal has a pronounced effect on the final protection capability and the amount of material removed. In "hot" structures and heat sink applications, oxide penetration and embrittlement may also be accelerated.

Most of the ablation data now being evaluated has been produced in freely expanding plasmajets. Free jets have certain limitations because of non-equilibrium flow, unmeasurable thermal gradients, constantly varying mass flow and frequent contamination with foreign particles or vapors. A carefully expanded jet is an essential requirement for any aerothermal test program.

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The major stumbling block to very high mass flow experimentation is the cost involved in handling large mass flow rates and in particular, heating these large masses of air. Large power installations are now becoming available and the lack of understanding associated with high mass flow and high pressure may soon be overcome. In view of reasonably high mass flow data produced in rocket exhaust there does not appear to be sufficient evidence for generally discrediting low mass flow-high enthalpy data.

Most low mass-flow, high-enthalpy systems operate at low pressure. Consequently, the Reynolds number is almost certainly in the laminar range. The question here is that of increasing mass flow of the system to increase Reynolds number - even to the extent of obtaining turbulent flow.

Chemistry

In this discussion two components are involved. First, the duplication of the proper chemical species in the environment and second, the prevention of contamination such as electrode material or other foreign matter. Comparisons of air plasmajet test and rocket exhausts test using the same enthalpies have compared favorably despite the very different chemical environment. In other tests the opposite is true (73) and the ablation rates are significantly different. However this is partially explained by the significant differences between the mass flow rates of the respective facilities. The removal of fragile char layers accelerated ablation rates.* In brief it is safe to recommend hot air jet exhausts.

*Air plasma jet and rocket exhaust tests at identical gas enthalpies may give comparable results, if the gas chemical effects are negligible. Ablative plastics that form surface chars are susceptible to changes in gas chemistry because of oxidative attack. Ablative plastics that form surface melts of an oxide type are generally relatively insensitive to gas chemical effects.

 [redacted]

This does not necessarily discount the usefulness of combustion sources and in particular suggests caution with regard to mass flow interpretations.

Time

Time is an essential component of any materials test. As a test parameter it is almost self explanatory; the reasons for its inclusion involves the elimination of test facilities which cannot supply sustained conditions for materials equilibrium and to specify it as a
related parameter to temperature, strength, weight and heat flux. In X
relation to these latter parameters there is no way of compromising time 6
or accelerating test conditions, as long as reliable scaling laws are 5
not established (79, 80). 0

5. APPLICATION TO FLIGHT REGIMES

The application of TPS's is determined by their adaptability to the environmental requirements of various vehicle systems. These requirements are comprised by the environment, previously defined by heat flux and time (Pt. I, 2b, and c), and by weight limitations as the primary criterion of lightweight design. As the capabilities of TPS are, likewise, represented in terms of $\dot{q}$, t and w, TPS's and vehicle systems can be compared on the basis of equal dimensions. This is accomplished in figure 11 by introducing lines of equal gross weight for various TPS's in the previously established heat flux-time diagram (fig. 4) identifying various flight regimes.

Even though this combined representation is somewhat crude, it conveys a clear picture of the applicability of TPS's, as well as of the missing capabilities, in terms of the heat flux-time requirements, which cannot be matched by presently available TPS. It is evident, e.g., that the present state-of-the-art in ablation systems is fairly adequate for the requirements of ballistic missile nose cones and drag-type (steep) space re-entry, i.e., very short time exposure to high and extreme heat fluxes. The time could even be extended by the use of transpiration systems. An adequate state-of-the-art is further existing in the field of short time exposure to intermediate heat fluxes, as primarily encountered in tactical missiles or missile defense systems, as well as in the low-heat flux-long time field, applying to steady state cruise flights up to about Mach 4-5, depending on altitude. A serious lack of capabilities is, however, apparent in the field of intermediate heat fluxes (15-250 Btu/ft^{2}sec) and prolonged times (4-60 min), representing an area of prime interest to NASA, as it includes the lift-type re-entry from space operations. This most outstanding problem area of TPS can be attacked from three sides: from the side of high-heat flux absorptive systems, by extending the time limitations; from the intermediate

[redacted]

capability systems, by increasing their time and heat flux capabilities; and from the side of the dissipation systems, by improving their thermal capability.

Of the absorptive systems, the low rate ("slow") ablation systems are most promising. Considerable work has been carried out with subliming plastics, particularly Teflon. Besides its thermal efficiency, sublimation is assumed to produce fairly clean ablation products, which is of particular concern to the integrity of the down-stream lift surfaces involved.

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Of the systems with intermediate thermal capabilities, the use of convective cooling, particularly the two-cycle systems, is technically feasible. While the practical application to limited "hot spot" components (leading edge) may be acceptable, its high weight requirements prohibit its application to large surfaces. Similar limitations are indicated for liquid film cooling. Extremely promising, however, is the gaseous film cooling or "boundary layer modification" system, as outlined in detail in Part II-2.

The thermal capabilities of insulation systems, finally, are far below the requirements of re-entry vehicles. While they will always remain attractive due to their time-independence, the required substantial improvements cannot be expected on the basis of present concepts and designs. Only fundamentally new approaches may lead to the required major break-through.

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PART II - APPLICATIONS TO NASA PROJECTS

1. DETERMINATION OF AREAS OF PRIME INTEREST TO NASA

After careful consideration, and in view of the existing comprehensive report of the Materials Advisory Board on the general subject (1), the Panel on Thermal Protection Systems decided to confine the further, more detailed treatment of TPS to those applications which are of prime interest to NASA.

For the purpose of this selection, a survey of all potential applications in aerospace engineering was made, which arrived at the following classification: X650

A. Super and Hypersonic Aircraft
   1. Low Altitude
   2. High Altitude

B. Manned Re-entry
   1. From Orbital Operations
   2. From Hyperbolic Path
   3. Boost Glide

C. Unmanned Re-entry
   1. Ballistic
   2. From Orbital Operations
   3. From Hyperbolic Path
   4. Boost Glide

D. Maneuvering Missiles

E. Solid Propellant Rocket Motors
   1. Nozzles
   2. Casings

F. Temperature Control of Space Vehicles

G. Ascent Heating of Boosters

H. Power Plants for Space Vehicles

I. Cryogenic Tankage

The selection of the areas to be discussed was based on three criteria:

a. The extent to which the application is within the overall mission and objectives of NASA

[redacted]

b. The severity of the problem
c. The degree of up-to-date coverage in existing reports

The finally selected applications are underscored in the above listing. They are discussed in the following chapters, prepared by individual members of the ad hoc panel.

It appears appropriate at this point to emphasize, that the following discussions do not represent an engineering handbook, nor is it attempted to invent new TPS. The prime objective of this report is to define the significant problem areas and to indicate potential solutions and their implications on materials research and development.

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2. ORBITING AND INTERPLANETARY RE-ENTRY

a. Definition of the Environment

(1) Description of the Mission

One of the pressing problems facing NASA is that of providing a thermal-protection system for manned vehicles entering the Earth's atmosphere at parabolic velocity or about 36,000 feet per second. It is not implied that an optimum thermal-protection system for entry from circular satellite speed has been accomplished. Rather it is recognized that protection for satellite-speed entry is within the state of the art of ablative heat shields for certain classes of blunt vehicles. For more slender entry configurations, ablative materials in the area of stagnation points and coated refractory metals in other regions appear capable of coping with the heat fluxes and total heat loads. For the case of higher velocity entry, such as is associated with returning from a lunar or planetary expedition, the ability to provide a workable thermal-protection system is not assured. At 36,000 feet per second the maximum heat flux due to convective heating is at least twice that at 25,000 feet per second. In addition, a new heating phenomenon has come into play, the radiative heating from the hot gas cap between the vehicle nose and the detached bow wave. This radiative heating rate is a function of the vehicle W/C_DA, the radius at the stagnation point, and the flight trajectory and under not unreasonable circumstances, can be shown to be equal to or to exceed the convective heating rate. It is these considerations that lead us to emphasize the thermal-protection requirements of the parabolic-velocity entry mission, recognizing that much materials research remains to be done in order to provide an optimum protection system for vehicles returning from earth-satellite orbits.

(2) Operational Characteristics

We shall limit our discussion here to entry of manned vehicles. There are several constraints on the entry trajectory that are imposed by specifying that the vehicle shall be manned. The vehicle deceleration must not exceed about 10g; multiple-pass grazing entries are not allowed due to the radiation hazard associated with numerous exposures and slow transit through the Earth's radiation belts; a point landing is required. To satisfy these requirements for a non-lifting vehicle and a single-pass entry requires a navigation system that can guide the vehicle into a narrow entry corridor with stringent limitations on altitude, flight-path angle, and time. By using a vehicle capable of producing lift, all of these requirements can be relaxed a considerable extent, but a single-pass entry and point landing, even for a lifting vehicle, demands an accurate, reliable navigation system and considerable on-board fuel for terminal guidance. Navigation accuracies can be still further relaxed by slowing the vehicle only from escape speed to satellite speed on the first pass. Application of a small amount of thrust will then establish a near-earth satellite orbit from which controlled entry and point landing can be accomplished following application of retrograde thrust.

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To describe the thermal environment associated with this entry mission, it will be helpful to define an entry corridor through which the vehicle must fly in order to accomplish atmospheric braking and final landing. The upper altitude of this entry corridor is fixed by an overshoot boundary. If the vehicle exceeds this altitude on the entry path, it will not be slowed sufficiently to enter in a single pass and a multiple-pass or grazing entry will result. The lower altitude of the entry corridor is fixed by an undershoot boundary. If the vehicle enters below the undershoot boundary, the deceleration imposed on the occupant will exceed the 10g limit. For a non-lifting vehicle, the width of the entry corridor is seven miles. With a constant L/D of 1/2 it is forty miles and for a constant L/D of 1 it is about fifty miles. By modulating lift and/or drag during entry, substantial additional widening of the corridor can be accomplished.

(3) Resulting Thermal Environment

The convective heating of a vehicle entering the atmosphere of the Earth or of other planets has been treated analytically by Chapman in NASA Technical Report R-55 "An Analysis of the Corridor and Guidance Requirements for Supercircular Entry into Planetary Atmospheres." The analytical treatment is based on laminar continuum flow, chemical equilibrium in the boundary layer, and a cold, noncatalytic wall. The assumption of laminar flow in the continuum regime is believed valid at the stagnation point near the region of maximum convective heating. The flow within the boundary layer is calculated to be in a state where we can expect finite recombination rates which should reduce the heat transfer from the calculated value based on equilibrium dissociation. The wall reactions are unknown but if the wall is cooled by an ablative or transpiration system, the chemical activity and recombination at the surface should not introduce serious errors.

Simplified expressions have been obtained for the maximum stagnation point convective heating rate and total convective heat absorbed for entry to the Earth as follows:

$$\dot{q}_c = \frac{19\left(C_q \bar{V}_E\right)^2 \sqrt{W/C_D A R}\, \sqrt{G_{max}}}{\left[1 + (L/D)^2\right]^{1/4}}$$

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$$Q_1 = \frac{8{,}800\left[1 + (L/D)^2\right]^{1/4} \sqrt{W/C_D A R}}{C_Q \sqrt{G_{max}}}$$

where $\dot{q}_c$ - maximum convective heat flux at stagnation point, Btu/sec ft^2

C_q - a constant of order unity tabulated in NASA TR R-55

$\bar{V}_E$ - entry velocity, non-dimensionalized by dividing by local circular satellite velocity, $\bar{V}_E = \sqrt{2}$ at parabolic velocity

L/D - aerodynamic lift-drag ratio

W - vehicle weight, lbs

C_D - vehicle drag coefficient

A - reference area of vehicle on which C_D is based, ft^2

R - vehicle nose radius at stagnation point, ft

G_{max} maximum deceleration in earth sea-level g's

Q_1 - total convective heat at stagnation point in decelerating from parabolic velocity to local circular satellite velocity

C_Q - a constant of order unity tabulated in NASA TR R-55

Inspection of these equations indicates that for entry to the Earth's atmosphere at parabolic velocity with a 10g limit in acceleration,

[redacted]

the maximum convective heat flux is reduced by operating at the lowest possible W/C_DAR and the highest possible L/D. On the other hand, operating at high L/D increases the total convective heat load by increasing the flight time at which the heating rate is high. A coupling also exists between L/D and W/C_DA in that an increase in L/D is usually accompanied by a decrease in C_DA and a resultant increase in W/C_DA. As a result of this coupling, and for entry at a constant L/D, increasing the lift-drag ratio at the 10g limit undershoot boundary of the entry corridor, leads to an increase in both heating rate and total heating at the stagnation point. For L/D of 1/2 or less, this effect is not particularly strong. X650

The phenomenon of aerodynamic heating by radiation of the hot gas cap between the vehicle nose and the bow shock wave is a new problem that assumes importance for manned entry vehicles as the flight speed exceeds about 32,000 feet per second. Our knowledge of the magnitude of the radiation energy from the hot gas cap is far from satisfactory. Experimental data are available from ballistic ranges at speeds up to about 26,000 feet per second. Shock-tube measurements have been made at lower velocities and Kivel and Bailey of the AVCO Research Laboratories have provided a theoretical method of extrapolating these results to parabolic velocity. In the following discussion the estimates of radiative heat transfer are based on the AVCO extrapolation but the results may well be in error by a factor of 2.

Consider first the case of a non-lifting vehicle on a single-pass undershoot entry from parabolic velocity, with a deceleration limit of 10g. The maximum radiative heating rate at the stagnation point, $\dot{q}_r$, is given approximately by the following expression:

$$\dot{q}_r \approx 0.01R\left(W/C_DA\right)^{1.8}$$

where the notation is as defined earlier. Compare this with the approximate equation given earlier for the stagnation point convective heating rate for the same zero-lift entry:

$$\dot{q}_c \approx 75\left(W/C_DAR\right)^{0.5}$$

Observe that the radiative heating varies directly with the nose radius and with approximately the square of W/C_DA. In contrast, the convective heat transfer varies inversely as the square root of the nose radius and as the square root of W/C_DA. For the case of a sphere, the convective and radiative heat fluxes are equal for a radius of about 6 feet and a W/A of 100. For the case of a 10-foot diameter spherically blunted cone cylinder wherein there is a coupling between nose radius R, and C_D, the convective and radiative rates become equal as the nose radius is increased to about 10 feet. Studies of time histories of the heating rates during entry reveal that the radiative and convective heating rates peak at nearly the same time in the trajectory, but that the period of radiative heating is only about one-third of that for the convective. For the case of the 10-foot diameter spherically blunted cylinder with a 10-foot nose radius, the total radiative heat load is about 25 percent of the convective.

In general, the effect of increasing the lift-drag ratio is to increase the radiative heating rates. At the undershoot boundary with a 10g deceleration limit, the radiative heat transfer at the stagnation point increases by an order of magnitude as the L/D is increased from 0.5 to 2 for a constant wing loading, W/A. A fairly low wing loading and modest L/D appear essential if high radiative heating rates are to be avoided at parabolic entry speeds.

Additional details on convective and radiative heating can be found in the published proceedings of the USAF/NASA Conference on Lifting Manned Hypervelocity and Reentry Vehicles.

(4) Secondary Environments

The preceding section has outlined the thermal environment associated with atmosphere entry. A thermal-protection system for a lunar or planetary vehicle must cope not only with this entry heating, it must provide the proper thermal environment for the vehicle during space-flight missions having durations varying from several days to many months. The problem of thermal-energy balance during cruising space flight will be discussed in a later section. It is important to note here that the surface coating must have the proper ratio of absorptivity-to-emissivity to provide this thermal balance. In addition, prolonged exposure to the space environment must not degrade the performance of the thermal-protection system during the period of intense convective and radiative heating associated with the terminal phase of the mission. For the case of an entry mission that slows to satellite speed in the first pass and then enters a parking orbit to achieve point landing capability, the thermal-protection system must retain or regain its ability to provide energy balance in orbit and must still have the capability to protect the vehicle against the rather severe aerodynamic heating associated with entry from satellite speeds. The thermal-protection system must also have the ability to sustain or be designed to avoid the aerodynamic pressure loads developed during the entry and must withstand the sizable loads associated with the vehicle deceleration.

b. Thermal Requirements

(1) Numerical Data on Heat Flux, Total Heat, etc.

In order to translate the environmental data given earlier, into actual requirements for a thermal-protection system, it is first necessary to select some specific vehicle configuration. The sizable increase in entry corridor width associated with a lifting entry and the concomitant relaxation of the requirement for extreme accuracy in terminal guidance, lead one to consider a vehicle having a maximum lift-drag ratio of about 1/2 for a manned lunar or interplanetary vehicle. This lift-drag ratio is sufficient to increase the entry corridor width to about forty miles, provide reasonable terminal range control, and the lift-drag ratio is sufficiently low to not seriously aggravate the heat flux and total heat load associated with convective and radiative heating.

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The vehicle selected for study has a W/C_DA of 100, a nose radius at the stagnation point of 2.5 feet, and is capable of flying at an L/D of 1/2 at the undershoot boundary or -1/2 at the overshoot boundary. The heating rates to be described are those associated with slowing the vehicle from parabolic velocity to satellite velocity in a single pass with a 10g limit on the deceleration. All heating rates and heat loads are for the stagnation point and the assumptions enumerated earlier are retained. Table 4 lists the important heating parameters. The numerical values are only approximate but are sufficiently accurate to provide an appreciation of the requirements of a thermal-protection system for this mission.

TABLE 4.- HEATING PARAMETERS FOR PARABOLIC RE-ENTRY

Trajectory	CONVECTIVE			RADIATIVE		
	$\dot{q}$, $\frac{\text{Btu}}{\text{sec/ft}^2}$	Q, $\frac{\text{Btu}}{\text{ft}^2}$	Time mins.	$\dot{q}$, $\frac{\text{Btu}}{\text{sec/ft}^2}$	Q, $\frac{\text{Btu}}{\text{ft}^2}$	Time mins.
Undershoot boundary L/D = 1/2	600	30,000	1.7	200	4000	0.7
Undershoot boundary L/D = 0	450	35,000	2.6	100	3000	1
Overshoot boundary L/D = 0	320	60,000	6.2	20	1200	2
Overshoot boundary L/D = -1/2	200	100,000	17	---	---	---

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Heat pulse is approximately triangular. Time given here is for a triangular pulse having a peak heating of $\dot{q}$ and total load of Q. In contrast, the maximum stagnation point heating rate of this same vehicle in a decay trajectory from satellite orbit is only 70 Btu/sec-ft^2, the total heat load is 50,000 Btu/ft^2, and the time of heating is of the order of 12 minutes.

The data of table 4 are plotted in figure 12 as stagnation point heat flux versus exposure time. Also shown are the thermal environments for several other missions as well as the areas of applicability of a number of thermal-protection systems. Admitting the approximations inherent in this analysis and recognizing that the numerical values apply to the single vehicle chosen for study, it is nevertheless postulated that the heating environment for this particular mission is more severe than any considered in the MAB report on Thermal Protection Systems and falls in an area where materials development will be required.

c. Potential Thermal-Protection Systems

(1) Discussion of Various Solutions

Of the thermal-protection systems that have been proposed and studied to date, it appears that systems based on ablation cooling, forced transpiration cooling, or film cooling, are the most promising candidates for this parabolic-entry mission. The radiation equilibrium temperatures are too high for any known materials; heat-sink and convection cooling systems appear impractical because of the long exposure times and large total heat loads.

(a) Ablation Cooling.- An ablation cooling system possesses features that make it look attractive as thermal protection against the convective heating flux of this entry mission. As pointed out in Part 1, the thermal capacity or effective heat of ablation against convective heat transfer varies directly with the driving enthalpy (difference between free-stream enthalpy and enthalpy at surface). Certain ablative materials with large blocking factors are calculated to have exceedingly large thermal capacities at the high enthalpies associated with flight speeds from 25,000 to 36,000 feet per second. Reliable experimental data are not available at enthalpies above about 9,000 Btu/lb., however, and there is a need for data at enthalpies of 25,000 Btu/lb to verify the extrapolation based on trends observed at the lower enthalpy levels. The performance of ablative materials at very low heating rates has not been established at any enthalpy level and although not pointed out in the earlier discussion, these entry trajectories subject the thermal-protection system to heat fluxes of 20 Btu/sec-ft^2 and below for periods of several minutes in the course of the entry maneuver. Finally, there is the question of degradation of ablative materials under prolonged exposure to the space environment and the tentative requirement that if a satellite parking orbit is used during the entry maneuver, the ablative material must protect against a second heating pulse associated with final entry from satellite speed.

Perhaps the greatest unknown and greatest concern about the use of an ablation cooling system for this mission is the degree of protection it will afford against radiative heat transfer. The feature of an ablation cooling system that makes it appear attractive as protection against convective heating is the large blocking effect of the vaporized materials that increases with increase in driving enthalpy. In the case of radi-
ative heating, no blocking effect can be expected unless the vaporized X
materials are highly absorbing or reflecting in the wave lengths of the 6
emitted gaseous radiation. Calculations have indicated that a substan- 5
tial reduction in total combined radiative and convective heat transfer 0
can be accomplished by injecting or transpiring a gas with a high absorption coefficient. Information is needed on the absorptive properties of gases from various ablative materials as well as much more information on the intensity and spectral distribution of the radiant energy from the hot gas cap. Most promising under simultaneous convective and radiative heat transfer are the char layer forming ablators and certain refractory materials which produce a significant reradiation from an opaque, high emissivity surface.

(b) Transpiration cooling.- Practically everything that has been noted above about ablation cooling systems is equally applicable to a transpiration cooled system. It is actually the transpiration effect inherent in an ablator that makes the system appear attractive against convective heating. The transpiration system offers a flexibility that the ablative system lacks in that the absorptive properties of the transpired gas may be selected to protect against the type of radiation which is predominant. Transpiration lacks the self-regulating features of an ablation system unless this can be incorporated in the system by clever design. The most serious criticism of a transpiration system is the demands it makes on porous or perforated materials. The capability of a transpiration system for protecting against both radiative and convective heating is apparent if the proper absorbing gas can be found; the system is adaptable to both high and low heat fluxes; the surface may be less subject to degradation from the space environment and more tolerant of a second heating cycle following a sojourn in a parking orbit than an ablative material. The potential capabilities of a transpiration-cooling system are such that it cannot be eliminated at this time although it poses demands on perforated materials that have not been satisfied to date for conditions much less critical than those described here.

(c) Film cooling.- As used herein, film cooling applies to a thermal-protection system in which a low-energy gas blanket is interposed between the vehicle wall and the high-energy gas stream.

Using helium as an example, an insulating gas film can be developed that flows along the vehicle surface and does not mix appreciably with the hot air in the boundary layer. Film-cooling systems have been analyzed by Ferri and Libby, and have been used experimentally to protect the nozzle and walls of high-temperature supersonic wind tunnels. A film-cooling system differs from a transpiration cooling system primarily in the manner in which the coolant is injected. For film cooling, injection is in the tangential direction from discrete slots; for transpiration cooling, injection is usually normal to the surface through a porous or perforated material. As noted before, a gas having a high absorption coefficient would be required to protect against radiative heat transfer and whether a single gas can be found that would be effective against both radiative and convective heating is a subject for further study.

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The most serious deficiency of a film-cooling system is the problem of providing a method of injecting the insulating gas in such a manner that it will not mix with the oncoming stream and will adhere to the surface downstream of the injection point. This is still further complicated when we consider non-symmetrical shapes and variable angle of attack as will be required for the entry mission we have defined. Perhaps the most logical application of a film-cooling system is near the stagnation point and on exposed control surfaces where the heating fluxes are a maximum. Film cooling appears attractive because of its high thermal efficiency but much research and development will be required before the practicability of this approach can be assured.

(2) Recommended Thermal-Protection Systems

As indicated in Section (b) on thermal environment, the heating history for this mission of atmosphere entry at parabolic speeds imposes requirements on the thermal-protection system that can only be satisfied by additional research and development. More precise information is required on the heating environment. The performance of various candidate systems and the properties of various materials must be further explored. It would be presumptuous to recommend a thermal-protection system for the parabolic speed entry mission at this time.

(3) Required Materials and Properties

The material requirements for thermal protection during atmospheric entry have been alluded to in the discussion of ablative systems, transpiration systems, and film-cooling systems.

For ablators we have the obvious requirement for the highest possible thermal capacity when exposed to combined convective and radiative heat fluxes ranging from 10 Btu/sec-ft^2 to over 1,000 Btu/sec-ft^2 at enthalpies from 4,000 to 25,000 Btu/lb. The surface material should

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not suffer degradation in performance when exposed to the space environment for periods of time ranging up to months. The requirement may be imposed that the material be suitable for protecting the vehicle during three distinct phases of the entry; the initial pass where speed is reduced from parabolic velocity to local satellite velocity; a period of time in a parking orbit; a final entry and landing from satellite speed. The possibility should not be overlooked of a combined thermal-protection system in which an ablative material is consumed in the initial pass, exposing a radiation-cooled structure capable of providing thermal-protection against the entry heating from satellite orbit. In addition to providing thermal protection the surface material must be capable of withstanding the aerodynamic pressure loads and the deceleration forces associated with the entry trajectory.

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For a transpiration-cooled system, a porous or perforated material must be developed having precise and controlled porosity and capable of withstanding reasonably high temperatures. The surface material should have a high strength-weight ratio and clogging of pores due to high temperatures and/or aerodynamic loads must be minimized. The porous surface must be protected against meteorite damage and erosion in space. A gas or vapor must be selected that affords maximum protection against both convective and radiative heating. Alternatively a two-stage system may be feasible wherein a gas able to cope with both radiative and convective heating is transpired at the early stages of the trajectory where both radiative and convective heating rates are high, followed by injection of a light gas that has been tailored to provide maximum protection against convective heat transfer.

The film-cooled system has many problems analogous with the transpiration-cooling system. In addition there is the problem of injecting the insulating gas in such a manner that it does not mix with the hot oncoming gases. The gas blanket must adhere to the curved surfaces downstream of the injection point and must provide protection over a range of angles of attack.

(4) Testing Requirements

The testing requirements fall into two distinct categories; those tests required to better define the heating environment, and those tests required to establish the ability of various materials and systems to provide protection against the aerodynamic heating.

With regard to the heating environment, our ability to calculate stagnation point convective heat transfer is believed to be quite good although experimental verification of the calculated rates would be highly desirable at the flight Mach number, Reynolds number, and enthalpy with real-gas effects included. Our calculative methods for predicting convective heating at points aft of the stagnation region, on the vehicle

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base, and on control surfaces are inadequate and experimental data and refined theoretical solutions are urgently needed. With regard to radiative heat transfer, experimental measurements and a reliable theory are both lacking. Experimental data at much lower speeds than those associated with the parabolic-speed entry, have indicated radiation intensity for nonequilibrium flows that are at least an order of magnitude higher than those for equilibrium conditions. Not only do we lack reliable information on the intensity of the radiation at parabolic velocity, our knowledge of the spectral distribution of the radiant energy is even more primitive. A considerable experimental and theoretical effort is underway on this problem and it is hoped that results will be forthcoming that will better define the radiative heating environment.

In addition to the above, a considerable amount of research is being conducted to discover vehicle shapes and entry trajectories that maximize the entry corridor width and minimize the entry heating. Initial results of one such analysis are described in an Addendum to this section.

Testing requirements for materials include measurement of thermal capacity of various ablative materials at the enthalpy levels and heat fluxes associated with the entry mission. The reaction of ablative materials to intense radiative heating must be established as well as the ablation performance at low heating rates. Ablative materials must be exposed to the vacuum, temperature, electromagnetic radiation, and particle bombardment of space for prolonged periods of time to determine their stability under the conditions to be experienced during space flight. The ability of an ablation material to restart the ablation process for a second heating pulse must be investigated if a parking orbit is employed during the entry, as the space environment may affect the ablation material. For temperature control in the space environment the ratio of solar radiation absorptivity to emissivity must be established for a variety of materials and coatings, and the stability of coatings must be determined.

For transpiration and film-cooling systems, additional theoretical and experimental effort is required to determine the effectiveness of various gases and vapors in protecting the vehicle against both convective and radiative heating. Development of methods for fabricating surfaces with uniform and controlled porosity from materials having a high strength-weight ratio at elevated temperatures must be accelerated if transpiration cooling is to receive consideration as a thermal-protection system. Methods of injecting a gas to satisfy the requirements of a film-cooling system should be studied.

d. Formulation of Recommendations

This study has indicated the severity of the heating problem associated with atmosphere entry at parabolic velocity. Study of a typical lifting vehicle has indicated maximum convective heating rates of 600 Btu/sec-ft^2, maximum radiative heating rates of 200 Btu/sec-ft^2 and maximum total stagnation point heat loads from initial entry to touchdown of 150,000 Btu/ft^2. The severity of this thermal environment places demands on the thermal protection system far in excess of any system in use or under advanced development at this date. Of the systems considered for protection of this vehicle, ablation cooling, transpiration cooling, and film cooling appear most promising. There are many unresolved problems in the use of any of these systems and only through intensive research by NASA, by other research organizations, and by industry can we expect to find a solution.

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e. Addendum

The preceding section has defined the thermal environment associated with atmosphere entry at parabolic velocity of a manned lifting vehicle having fixed geometry. Use of lift was considered necessary to widen the entry corridor and thus relax the accuracy requirements of the terminal guidance system. Widening of the entry corridor can also be accomplished by varying the drag of a non-lifting vehicle. The results of such a calculation are shown in table 5 for entry at parabolic velocity at the undershoot boundary with a 10g limit on deceleration.

TABLE 5.- WIDENING OF THE RE-ENTRY

CORRIDOR FOR DRAG VEHICLES

$\frac{(C_DA)_{max}}{(C_DA)_{min}}$	Corridor width miles	$\frac{(\dot{q}_c) \text{ for } (C_DA)_{max}}{(\dot{q}_c) \text{ for } (C_DA)_{min}}$	$\frac{(\dot{q}_r) \text{ for } (C_DA)_{max}}{(\dot{q}_r) \text{ for } (C_DA)_{min}}$
1	7	1	1
5	12	.58	-----
10	23	.48	-----
20	30	.41	.016
50	38	.34	-----

[redacted]

As a numerical example consider a non-lifting Mercury-type capsule having a W/C_DA of 63. By extending a skirt, an umbrella, or some such adjustable device, assume that a frontal area ratio (brake extended/brake retracted) of 20 can be achieved. With this range of area ratios available for maneuvering, the following improvement over a fixed geometry nonlifting capsule can be achieved:

(1) Corridor width increased from 7 miles to 30 miles
(2) Peak convective heating rate reduced from 184 Btu/sec-ft^2 to 75 Btu/sec-ft^2
(3) Peak radiative heating rate reduced from 189 Btu/sec-ft^2 to 3 Btu/sec-ft^2
(4) Maximum radiation equilibrium temperature with $\epsilon = 0.90$ reduced from 4,950° F to 3,150° F

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It is obvious that an approach of this type has several attractive features. Within the accuracy of our calculations, the problem of radiative heating can be eliminated. The heat fluxes are sufficiently low that one can now consider a radiation-cooled structure. The success or failure of this modulated drag approach, hinges on the development of a flexible, nearly impervious material for the drag brake that is capable of operating over the required range of temperatures, is light in weight, and has sufficient strength to withstand the aerodynamic pressures and the deceleration loads.

3. THERMAL CONTROL IN SPACE VEHICLES

a. Thermal Environment and Control

Except for periods of exit and entry through planetary atmospheres, most space vehicles will operate in environments of such low pressure that radiation will be the predominant mode of heat transfer between the vehicle and its surroundings. The sources of energy will be such that the bulk of the radiation affecting the heat balance of the vehicle will fall in the range of wavelengths between 0.2μ and 50μ. With one major exception, Saturn with its rings, and some minor exceptions, such as meteors and meteoric showers, the source-vehicle geometries cause the radiation to fall into three general categories:

(1) Radiation from distant objects (parallel rays)
(2) Emission of radiation by large nearby spheres
(3) Reflection by large nearby spheres of radiation from distant objects

The geometry of the vehicle and its relation to the energy source (generally varying with time) determine the magnitude of the total irradiation. Equally important are the spectral-energy distributions of the sources and the spectral-radiation characteristics of the external surface which determine the abs[redacted]d affect the reradiation

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from the vehicle surface. The other important factor in the heat balance is the internal power dissipation. These factors - the absorbed energy, reradiation, and power dissipation, together with internal conduction and radiation effects - determine the temperature boundary conditions upon which the internal thermal design is based. The internal temperature environment will depend on the proper choice and control of "external surface" finishes which are of paramount importance in fixing the external temperature boundary conditions. Reasonably stable finishes, for which the radiation characteristics and the effect of prolonged exposure to the environment on these characteristics are known, must be used. Much of the required basic information on surface effects is unavailable; X
however, programs are being conducted in several places to obtain the 6
required data. 5
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It is necessary to dissipate power on the average through some heat-flow path to a sink at some temperature while maintaining the power-generating units and other components within a given temperature range. Although it is feasible in some special cases to use power as a means of control, it is generally required that the other two factors - the heat-flow path and the sink temperature - be either actively or passively controlled to yield the desired regulation of temperature and power dissipation.

b. Uncertainties

All of the factors entering into the heat balance, externally and internally, are uncertain to some extent because of variations in initial conditions, tolerances on nominal factors, lack of complete information, and inexact analytical methods. The uncertainties in surface-finish characteristics, external heat sources, and internal heat-flow paths must be evaluated before temperature controls can be designed.

(1) Prediction of α and ϵ

The ideal set of surface-finish data would include the dependence of the spectral radiation characteristics on the angle of incidence (or angle of emission) and on the temperature of the surface. Since such extensive data are not available, the designer must estimate the necessary absorptances and emittances from theory or from normal spectral, total normal or, sometimes, total hemispherical experimental data.* The Hagen-Rubens equation can be used with a fair degree of accuracy to approximate the reflectivity of bare polished metals for long-wave energy at normal incidence; theoretical relations given by Eckert (13) can be

*In accordance with present day recommended definitions, the suffix "-ance" refers to the radiating properties of a body compared to a black body, whereas terms ending in "-ivity" refer to the same ratio for opaque materials having an optically smooth surface composed of the same material.

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used to estimate the necessary total hemispherical from the total normal radiation characteristics; and the angular dependence of the radiation characteristics can be estimated from Fresnel's formulas and electromagnetic theory (14), provided the material meets certain stringent requirements. However, the theories are incomplete, especially in the short-wave region where most of the direct and reflected solar energy lies. Furthermore, the radiation characteristics and their angular dependence are not clearly defined for rough or thinly coated materials and thus not easily estimated. For these reasons, the theoretical values are highly uncertain.

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Even though no materials obey Lambert's cosine law exactly, some follow it closely enough to be regarded as perfectly diffuse reflectors and emitters. Many dielectrics fall into this category (15). On the other hand, no materials are perfectly specular reflectors, although many, notably bare metals, have a strong angular dependency for reflection and emission. If such a material were to be used on a cylindrical satellite with an attitude constantly vertical relative to the earth, the determination of the absorptances of the sides of the cylinder for earth emission and reflection should take the angular dependency into account. All of the energy from the earth would be incident on the cylinder at angles less than the angle with a sine equal to the ratio of the earth's radius to the radial distance of the cylinder from the center of the earth. Using data given by Eckert (13), it was found that the absorptivity for earth emission on a vertical polished nickel plate or cylinder at an altitude of 200 mi is 12 percent higher than the total hemispherical emissivity and 22 percent higher than the total normal emissivity (this calculation was done under the assumption that the absorptivity for earth emission is equal to the room-temperature emissivity). The influence of angular dependency would be of less importance for spheres or horizontal cylinders.

(2) Environmental

Additional errors are introduced because of unknowns regarding:

Absorptance for earth emission and the emittance at the temperature of the vehicle surface

Angular dependency of the two terms

Solar and albedo absorptances and their angular dependencies

Even if the surface is a diffuse absorber for which spectral absorptances are available, it is still not possible to integrate to obtain the absorptances for earth emission and earth-reflected solar energy because exact spectral-radiation characteristics are seldom measured beyond 25μ, about

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25 percent of the energy from a black-body at 250° K (the model used to approximate earth emission) lies beyond 25μ. If measurements were made to 50μ, 98 percent of the energy would be included.

(3) Miscellaneous

(a) Experimental and Manufacturing

Additional uncertainties in the radiation characteristics arise from experimental inaccuracies and a lack of control of the specimen. Experimental errors can easily produce an error in a measured emittance or absorptance of 0.03; if spectral data are integrated to obtain total characteristics, the error may be 0.02 (16). Such errors are especially serious for highly polished metals. If the solar absorptance and infrared emittance are obtained independently, the resulting α_s/ϵ ratio can easily be in error by as much as a factor of two. This fact suggests that a direct measurement of α_s/ϵ would be desirable, since satellite temperatures are so sensitive to the ratio.

Surface irregularities, coating thicknesses, the extent of oxidation, and other physical aspects can alter the radiation characterisitics greatly but are often difficult to define or control. Sample differences of this nature are very common, and inadequate sample descriptions limit the usefulness of a great deal of reported data for preliminary selection of surface materials. Measurements of characteristics of samples taken directly from manufacturing stock are generally necessary to insure proper representation. Even then, poor quality control can allow wide variations in the surface finish. Thus, manufacturing quality control is in itself a problem.

(b) Pre- and Post-Launch

The influence of the pre-launch and post-launch satellite environments makes surface radiation characteristics even more uncertain. The materials will have to withstand the normal manufacturing and handling processes; corrosive atmospheres before launch; extreme temperatures, chemical reaction, and erosion during ascent (especially if exposed, as in the case of Agena surfaces); and temperature cycling, particle bombardment, chemical reaction, sublimation, and radiation damage in a nearly perfect vacuum in orbit. For example, the nature and extent of micrometeorite bombardment are so ill-defined that, usually, only an upper limit on the rate of erosion is predicted. Whipple estimates that as much as 2×10^{-13} g/cm²-sec might erode from an aluminum surface (17). The rate of erosion as such is of interest when thin coatings are used on the satellite. If thick materials are used, however, it is more important to know whether the surface is pitted, polished, or plated. Such information is difficult to predict theoretically, but studies are being made under simulated orbit conditions in an attempt to estimate the extent and nature of the damage.

In a similar fashion, sublimation of surface materials will affect the surface finish. Heller, using Langmuir's relationship between vapor pressure and sublimation rates, has estimated that materials with a vapor pressure greater than 10^{-8} mm of Hg at orbit temperatures would be subject to serious damage, even when the influence of external excitations is neglected (18). When the removal of surface atoms and molecules (atomic and molecular sputtering) by the corpuscular radiation from the sun is added and the chemical reactions with ions and reactive atmospheric gases in orbit are considered, the removal rates are even more accentuated. Whipple estimated that surface erosion by corpuscular radiation from the sun and by gases of the extended solar corona would be approximately equal to erosion by meteoric dust (17). Furthermore, these removal rates are heightened if the lattice bonds are weakened by high-energy electromagnetic radiation.

Finally, damage due to Van Allen radiation, cosmic radiation, and any internal sources of nuclear radiation must be considered. Interactions with these types of radiation may adversely affect the surface by causing lattice defects which might increase the electrical resistivity which would in turn increase the emissivity. Obviously, the overall effect of the orbit environment is difficult either to predict or to simulate at this time. Early satellites (19, 20) and spacecraft should provide more of the information necessary for the design of long-lifetime vehicles. Furthermore, experimental programs such as the one described in the discussion on surface finishes will go far in minimizing the uncertainties in surface-finish radiation characteristics.

c. Surface Finishes

(1) Temperature Control

Surface temperatures on an earth satellite are largely determined by the α_s/ϵ ratio and ϵ where α_s = surface absorptance for solar radiation and ϵ = infrared emittance. From this, it is clear that careful control of α_s/ϵ and ϵ is required. It is usually necessary to determine the radiation characteristics for each new surface material of interest since these characteristics are a very pronounced function of surface condition and, consequently, will vary from sample to sample of a nominally identical material. Although the characteristics for many materials are available in the literature (21), a comparison of the values listed for the same material quickly indicates the generally unreliability of the literature values of emittance for thermal design use. For these reasons, we must develop the ability to carefully control α_s and ϵ of the interior and exterior surfaces of space vehicles.

One ultimate aim is that suggested in (22), i.e., the development of the four stable standard surfaces illustrated in figure 13 where α_λ is the monochromatic absorptance, and λ is the radiation wavelength.

These surfaces would either be stable throughout their lifetime or have a predictable change in their properties due to the effect of their environment. With these four surfaces, it is possible to construct mosaics with nearly any desired set of characteristics. For example, consider the frustum of a right circular cone striped with two of these standard surfaces as shown in figure 14.

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If the widths of the stripes are small and there are good thermal conduction paths between adjacent stripes, the difference in temperature between any two adjacent stripes can then be shown as negligible. A heat balance will show that the surface of the cone behaves as if it had an α_S and an ϵ which are the area-weighted averages of α_S and the ϵ of the two distinct striping surfaces. These four standard surfaces would be developed primarily as exterior surfaces.

A parallel effort should develop a material with thermal radiation characteristics that would enable the skin temperature of a spacecraft to remain sensibly constant despite orbital variations in the amount and kind of energy incident upon the surface.

Experimental work at one company is to begin soon on a ferromagnetic material with an emissivity that may be reasonably expected to exhibit a pronounced increase at the Curie temperature. If this expectation is justified by the experimentation, it appears possible that a constant temperature surface may be achieved.

<u>(2) Behavior During Pre-Launch, Launch, Orbit and Entry</u>

The environments to which these surfaces will be exposed are unusually severe and, in some areas, little is known about them. It must be realized that not all surfaces will be exposed to the same environment. The surfaces can be divided roughly into groups as follows:

<u>Non-Entry Surfaces</u> - This group includes surfaces that will not be used as radiation control surfaces during or after entry, namely:

(1) Exterior during ascent and in space

(2) Exterior in space only; interior during ascent

<u>Entry Surfaces</u> - These surfaces are those to be used as radiation control surfaces during or after entry.

Some control surfaces developed primarily for non-entry use may be usable as entry surfaces without any modification. It is also possible that, in some applications, vehicles may be resurfaced after entry. In the case of polished metal surfaces and surfaces of film-forming materials such as paint, this resurfacing might be easily performed. If, on the other hand, the reapplication of some anodic or electrodeposited surface were desired, it could become necessary to remove physically sections of skin from the vehicle.

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The environmental conditions themselves are conveniently divided into four groups. The first in time can be termed the pre-launch environment. In this environment should be included manufacturing processes and ground handling, and any scheduled test firings. Either the surface will tolerate certain amounts of abuse during manufacturing and ground handling, or special precautions must be taken to assure proper protection. The pre-launch environment also includes humidity, salt fog, and other sources of atmospheric corrosion. It should be noted that surfaces of a space vehicle are not normally subjected to atmospheric corrosion for extended periods of time. The fact, for example, that some porcelain enamels developed for use on aluminum have poor weathering qualities does not exclude their use on space vehicles. If necessary, protective coverings, such as strippable paints, can be used to protect surfaces during the pre-launch environment.

Both the ascent and entry environments include high heat rates and temperatures up to the structural limit of the materials being used, high aerodynamic shear forces, vibration, shock, and ionized and dissociated gases.

Some aspects of the third environment, space, are not yet clearly defined. An extensive effort is being made to gain increased knowledge of condition in space through the satellite programs and laboratory research. The space environment includes skin temperatures in the range of -150° to +300° F; vacuum; high ultraviolet intensities; Van Allen and cosmic radiation; X, gamma, and beta radiation; Winckler radiation; micrometeorites; ionized and dissociated atmospheric gases; and high-energy protons from the sun. Sputtering is caused by both gas particles and high-energy protons.

The experimental approach to the overall program may be exemplified by a description of the program in the Satellite Systems, Missiles and Space Division, of Lockheed Aircraft Corporation.

(3) Measurement of Emittance

Emittance measurements are being performed in three ways (23, 24, 25). Spectral or monochromatic-emittance is being measured by use of an integrating sphere and a Cary spectrophotometer in the short wavelength range

(0.4 to 1.8), and a hohlraum reflectometer and Perkin-Elmer spectrophotometer in the long wavelength range (1.8 to 25). Actually reflectance is measured, and emittance is computed from the basic definitions. An effort is being made to extend the range of the last method to 50μ which is desirable since approximately 25 percent of the radiation emitted by the earth is believed to be of wavelengths greater than 25μ. The short wavelength range will soon be extended to 0.22μ.

Total hemispherical emittance is being measured directly by a calorimetric method for temperatures between -200° and 2,000° F. The variation of monochromatic emittance α_λ and absorptance α_λ with temperature is being investigated as is the angular dependence of α_λ and ϵ_λ. A study of conductors and nonconductors to determine minimum film and plating thicknesses which may be employed to give values of α_λ and ϵ_λ which are not influenced by the substrate is under way. The determination of these minimum thicknesses will greatly simplify and improve careful control of α_s and ϵ when film-forming materials and metal plated surfaces are employed. Work is also going forward on surveying all present techniques and methods of measuring emittance. Emittance measurements are being made before and after subjecting a particular surface to a specific environment condition, in order to determine the effect of that condition on emittance.

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(4) Simulation Testing

Ascent heating is being simulated by radiant heating in the proper vacuum. The effects of aerodynamic shear and dynamic pressure on film-forming materials have been studied in flights of the X-7 supersonic test vehicle. These flights have also given information on heating effects.

Temperature cycling in a vacuum is being achieved with radiant heating. The present experiments simulate Agena vehicle orbits, some of which give much more severe cycling conditions than would be encountered by many space vehicles. The major effects of this condition are sublimation and the possible failure of surface films, such as paints, due to thermal stress and fatigue. Generally, any surface used for thermal control of a space vehicle with a lifetime greater than a few weeks should be composed of substances with low vapor pressure.

Accelerated tests of the effects of extended exposure to solar ultraviolet radiation in space on thermal radiation characteristics and surface stability are underway. It is expected that the effects of ultraviolet radiation on thermal radiation characteristics will be small; a probable exception is found in the case of film-forming materials. Several effects degrading both to the mechanical properties of the film and to its radiation characteristics are likely to occur on organic polymers. For paint, in general, some discoloration is also likely.

The effects of sputtering due to ionized and dissociated atmospheric gases are under experimental study (26). The combined effects of sputtering due to atmospheric particles and to high-energy protons from the sun is to be tested in a forthcoming Agena vehicle flight. It is felt that such effects on the thermal radiation characteristics of surfaces will be small for lifetimes less than one year.

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Information gained from previous U.S. satellites and from Russian sources on the intensities of micrometeorite bombardment is being examined.

Evaluation of high-energy space radiation data - Van Allen radiation; cosmic radiation; X, beta, and gamma radiation; and Winckler radiation - is to be performed as a part of this program. This evaluation will be made in conjunction with (1) the prediction of expected effects of these types of radiation on thermal radiation characteristics and surface stability and, (2) the subjecting of proposed surface coatings to effective radiation doses (27, 28). A study of the effects of nuclear radiation will also be carried out by the group evaluating the effects of high-energy radiation.

(5) Materials

It will be necessary to determine whether the selected surface finishes can safely undergo the vibration and shock of ascent and exposures to the fumes of the propellant used in the Agena vehicle program.

The major effort of the program thus far has been the evaluation of presently available surfaces, many of which exhibit some of the desired characteristics of the four stable, standard, and exterior surfaces.

Some black finishes, e.g., the Dow 9 surface on magnesium-thorium alloy, and black chromium electroplate, act as flat absorbers. Commercially available silicone-based black paints with carbon black or graphite pigmentation appear to be promising, possessing good resistance to ascent heating conditions. The development of soluble silicate, titanium ester, and silicone ester paint vehicles with suitable black pigments, are being investigated as possibilities showing definite promise (29, 30).

Some polished metals and even lightly oxidized metals act as solar absorbers because their ratio of α_s to ϵ is high. Aluminum with normal atmospheric oxidation has such surface; so do the precious metals. Some advantages over vacuum deposition are apparently enjoyed by an electrodeposition; a greater range of thicknesses may be achieved with more control than is normally possible with the vacuum process. Processes for depositing metals on magnesium in a vacuum or by electrolysis require further improvement. Other very effective solar absorbers are Tabor coated-metal surfaces (31, 32).

Paints pigmented with metals are the best examples of the flat reflector: α_s and ϵ are about 0.3.

Typical solar reflectors are magnesium oxide and titanium dioxide powders, or paints containing these pigments. Other pigments, such as strontium titanate, are being investigated. The same paint vehicles being considered in connection with flat absorbers are under consideration with X
white pigments, as solar reflectors. Consideration is being given to 6
"Ematal white," and anodic process for the deposition of white pigments 5
on aluminum. Polished metals with transparent coatings such as lacquer, 0
silicon monoxide, or Mylar can also be considered as solar reflectors.

It is possible that porcelain enamels or arc-sprayed coatings could be developed that would possess the characteristics of each of the four standard exterior surfaces. Difficulty is encountered, however, in attempting to apply these surfaces to aluminum and magnesium without affecting the mechanical properties of the substrate. Vitreous enamels show more promise than arc-sprayed coatings, and the development of both a flat absorber (black enamel) and a solar reflector (white enamel) for space vehicle application is possible; this course is being pursued (29). Porcelain enamels, which by definition must be fired above 800° F., are unsuitable for light metals where a maximum of mechanical properties must be maintained.

Ablative materials are under consideration as protective coatings during the pre-launch environment (to be removed during ascent) and as protective coatings through ascent (to be removed by sublimation in space). Strippable paints are being developed for protection during the pre-launch environment. A vinyl-base coating seems to be the most suitable commercially available material.

There are several problems associated with the production process itself (33, 34). Many surfaces with desirable characteristics require careful handling during manufacturing. The previously mentioned strippable coatings and ablators may be mandatory in some instances for this reason. In the case of surfaces which consist of an opaque film of one material over another, application must be controlled so that the proper minimum film thickness is maintained. This thickness is the minimum which may be used if the emitting characteristics of the surface are to be independent of the substrate. The Tabor interference coatings, and other special surfaces, require extremely careful control of the film thickness.

Film thickness guages, such as those employing an eddy-current skin effect principle, will be necessary for production control. The relative merits of several of these instruments are being weighed at the present time.

d. Conclusions

The above discussion indicates there is a great lack of information concerning the basic characteristics of and environmental effects on space-age materials. There is, therefore, need for the compilation of a design handbook or a series of handbooks. Complete information should be available on α_s, spectral emittance, and total hemispherical emittance at various temperatures, including the effect on these values of varying durations and severities of the various environments. Manufacturing instructions, methods of application and control, and any special handling instructions, such as the necessity for the use of a specific strippable paint, should be set down in detail. This handbook should be cross-indexed according to type of surface (i.e., paint, vacuum-deposited metal), α_s/ϵ, ϵ, and classification of surface by usage (i.e., non-entry - interior only), including the duration and intensity of the exposure to specific environments that the particular surface should be expected to withstand. This handbook should indicate the kind of material needed to produce surfaces having optical characteristics within specified ranges.

With the proposed design handbook, it would be possible for a designer, with little or no experimental verification, to select successful thermal control surface coatings for any given application. Reliability would thus be enhanced, and possible costly and time-consuming design delays avoided.

Successful achievement of optimum thermal control of all space vehicles will be long in coming unless the rate of effort on programs as described above can be greatly increased.

4. ROCKET NOZZLES

a. Definition of Environment

(1) Description of Components

The nozzle components of large, solid propellant rocket motors are generally designed to a convergent-divergent deLaval contour, with geometrical variations from conical to "bell" type contours, dependent upon the aero- and thermodynamic requirements of the particular motor. The

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large motors of the POLARIS class or larger utilize multiple nozzle configurations, both to limit the size of the nozzle to configurations capable of being produced by present techniques and manufacturing capabilities, and to permit reasonably simple, lightweight thrust vector control mechanisms.

The thrust vector control systems, which may be considered as part of the nozzle assembly, are commonly of two types: external "jetavator" flame deflectors, or internal swivel mechanisms by which a section of the nozzle is moved to deflect the propellant gases.

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(2) Operational Characteristics

The duration of firing of the present generation of large, solid propellant motors ranges from approximately 30 seconds up to 90 seconds. With the advent of segmented motors and greater range and load requirements, the probability is great that firing durations up to 120 or 180 seconds will be required.

The combustion temperature of production motors using aluminized polyurethane propellants is approximately 5,400° F. New developments in propellant chemistry have extended this temperature to approximately 6,300° F for the more advanced systems, and an ultimate temperature greater than 8,000° F may be considered a not too distant prospect, based on projected propellant developments.

The stress field which obtains in solid propellant motor nozzles is primarily due to the chamber pressure and to thermal stresses. Secondarily, vibration and "G" loading contribute to the stress field. Chamber pressures may range from 200 psig to 2,000 psig, according to the system. The thermal stresses are a function of the thermal and mechanical properties of the nozzle material and may be varied considerably by judicious design.

The propellant gas flow is supersonic in the nozzle throat, and may contain entrained liquid or solid metallic particles (aluminum) in addition to the combustion products of the propellant. The various components of the propellant gases are generally chemically reactive and may be either oxidizing or reducing in character, depending upon the design of the motor.

(3) Resulting Thermal Environment

Based on the time, temperature, stress, etc., conditions discussed above, the resulting thermal environment is characterized by:

(a) Very steep thermal gradients during the first few milliseconds of firing.

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(b) High heat flux as the nozzle wall comes up to "steady state" temperature.

(c) Chemical reaction and phase changes of the propellant gas and entrained particulate matter (aluminum).

(d) Free stream gas temperatures up to 6,300° F.

(4) Secondary Environments

The secondary environments of significance would include:

(a) the high mass flow rate of turbulent gases along the nozzle wall, which induces severe scrubbing or erosive action in the vicinity of the throat of the nozzle;

(b) the deposition and freezing of aluminum particles on the nozzle wall, particularly where wall temperatures are below the boiling temperature of aluminum, or below the freezing temperature of aluminum oxide;

(c) diffusion and/or reaction phenomena which may seriously alter the composition or properties of the nozzle material.

b. Thermal Requirements

The thermal requirements of any nozzle assembly would be directly dependent upon the thermal protection system to be used. Obviously an ablative system requires that the nozzle material would have a low melting temperature compared to the propellant gas temperature, and that thermal conductivity of the material be low. Good mechanical strength of ablative systems would not be a prime requirement since mechanical loads may be supported by structural shells, etc. A heat sink or "brute force" thermal protection system would require that the nozzle materials have high melting temperatures, good strength at temperature, and other properties not required of ablative systems. In between these two extreme types of thermal protection systems lie systems with varying thermal requirements.

It is therefore considered best to discuss the thermal requirements and numerical data of nozzle thermal protection systems, as they exist for each system.

Any review of thermal protection systems for solid propellant motor nozzles would reveal that for each combination of cooling systems, motor design, propellant characteristics, and operating specifications the thermal requirements and subsequent heat-fluxes and total effective heat capacities are singularly different. It would seem impossible to present information on heat fluxes or other heat transfer characteristics of a specific thermal protection system for a particular motor design which would be useable in some other design. Actually, variation in one parameter of engine operation such as gas pressure or temperature may render a previously successful thermal protection system invalid. The values of $\dot{q}$ and Q^* (defined below) are so interdependent for a given thermal protection system in a particular motor, that citing numbers out of the context of the complete motor and propellant specifications is less than academic.

The heat flux across the gas - solid interface of a rocket nozzle, X
normal to the surface, is generally referred to as $\dot{q}$, and is ordinarily 6
expressed in terms of Btu/ft^2/sec. This quantity is a function of many 5
inherent physical characteristics of the gas and the solid and also of 0
the operating parameters of the motor and nozzle cooling system. However, one of the most important variables in determining $\dot{q}$ is the difference between the stagnation gas temperature and the temperature of the solid surface (wall). With all other conditions remaining constant, the heat flux would be greatest where ΔT is greatest and would approach zero as ΔT approaches zero. It can be seen that for any constant wall temperature, $\dot{q}$ is independent of the particular thermal protection system used, or to phrase it differently, $\dot{q}$ may be maintained at any value between the maximum and minimum possible values, by maintaining a predetermined wall temperature. This implies that for any rocket motor cooling system, $\dot{q}$ may be considered to be also independent of the gas temperature if the thermal protection system may be varied to maintain the desired ΔT.

The generally accepted definition for Q^* is: Q^* = the effective thermal absorption capacity of a system per pound of the active part of a TPS, or, in the present case per unit mass of coolant. Both the mass of the coolant and the inert components associated with the cooling system should be ultimately considered in evaluating particular thermal protection systems. Generally, Q^* implies the consumption of a coolant during the cooling process and is not a primary consideration in radiation equilibrium cooled ("uncooled") solid motor nozzles in which no coolant material is consumed.

To date, very little firm thermal data have been acquired on firings of cooled solid propellant motor nozzles. However, considerable theoretical study effort has been maintained on various cooling methods and their theoretical cooling efficiencies. These studies include some qualitative statements of the expected $\dot{q}$ and Q^* values for the respective systems.

Theoretical expression of the thermal energy transfer from a flowing hot gas to the walls (and subsequently to the structure) of a rocket nozzle involves correlation between the physical, thermodynamic and hydrodynamic properties of the gas, and their effect upon the nozzle materials. Many investigators have derived expressions for various heat-transfer models. These expressions include the dimensionless Nusselt, Prandtl and Reynolds numbers for the flowing gases, which imply knowledge of the many physical properties of the gases, and also include knowledge of the various physical properties of the solid nozzle wall and structure, <u>all versus temperature</u>.

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The validity and accuracy of these theoretical expressions is sometimes questionable, even for simple models involving single species of gases at relatively low flow rates and temperatures, because of the difficulties in experimentally determining quantitative values for the various thermal and physical properties of the gases and solids involved, at the temperature of interest. For complex configurations such as de Laval and plug type nozzles, and for high velocity, very high temperature, multi-component gases, the theoretical expressions of heat transfer, including those for $\dot{q}$ and Q^* are no more than useful approximations. These expressions are made even less precise when continued (and sometimes unknown) chemical reactions are taking place within the propellant gases and between the nozzle material and the gases. The addition of solid (or liquid) particles of aluminum, boron, lithium, or other materials to the propellant stream serves to reduce the value of theoretical expressions of heat transfer to "educated guesses".

c. Potential Thermal Protection Systems

(1) Discussion of Various Solutions

Over the past two decades many methods of cooling rocket nozzles have been proposed and investigated. The efficacy of these methods is strongly dependent on the rocket system and mission desired, but based on these studies optimum systems for specific missiles may be developed. Recommended candidate cooling systems to be discussed are:

(a) Transpiration cooled
(b) Film cooled
(c) Evaporation cooled (nucleate boiling)
(d) Chemical reaction cooled
(e) Propellant cooled
(f) Ablative cooled

Let us consider these systems separately.

(2) Recommended Thermal Protection Systems

(a) Transpiration cooling

A great deal of theoretical and experimental work has been done in the field of transpiration or "sweat" cooling. Work done at AGC has developed data sufficient to allow determination of heat-transfer coefficients, wall temperatures and wall thicknesses as a function of coolant flow rates (35, 36).

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The transpiration system has a distinct advantage over other types of nozzle cooling because the mass flow of the coolant in the nozzle is in the opposite direction of the heat flux. It is, therefore, not necessary to attempt to maintain a boundary layer in the nozzle entrance or throat section where optimization of design is already a difficult problem. Experimental work on such widely different porous materials as copper, Nickel-Moly-Iron alloy (37), stainless steel and graphite have shown that wall temperatures are nearly independent of the thermal properties of the nozzle wall material. Therefore, sintered porous molybdenum, or any other suitable flame barrier material, could be used with little regard for its heat-transfer characteristics.

Because of the intimate surface contact between the coolant and the nozzle material (high specific surface of the porous material), cooling efficiencies are quite high and the thermal effects on surface areas exposed to the hot gas stream are of limited importance. Using air or nitrogen as a coolant, actual coolant flow rates of from 0.5 to 1 percent of the throat-flow-rates have given 1,500° F temperature drops between the nozzle wall and the main-stream at steady state conditions. By using helium or hydrogen as a coolant it should be possible to maintain more than 2,000° F temperature difference with 1/4 of 1 percent of the throat flow-rate if stability of coolant flow can be maintained. Experimental evidence shows that hydrogen is about 5 times as effective a coolant as is nitrogen for a given mass flow rate (36). The controlling factors are the differences in heat capacities of the gases and the higher velocity of the lighter gas within the porous material and at the interface (37) of the nozzle and coolant streams.

The principal difficulties with this system, when applied to aluminized propellants, are the erosion of the nozzle by aluminum oxide particles and the possibility of clogging the porous areas. A suitable mass flow through the nozzle wall should give some protection from this effect.

Transpiration experiments, where a dissociation of the coolant occurs, have shown the coolant flow to be unstable with a resultant increase in local temperature that is relatively uncontrollable (36, 38, 39) even though the heat absorption with dissociation is higher for a given mass flow. This is caused by the large decrease in the mass flow rate after the dissociation has taken place within the porous material. In addition, the flow of coolant circumnavigates a local hot spot, owing to the increase in pressure upon dissociation, which allows the hot spot to grow even larger. In order to make use of the heat of dissociation, it is necessary to assure that any dissociation occurs outside of the porous liner. This would necessitate a very sophisticated system requiring test work within closely restricted mensuration tolerances and combustion temperatures. Currently, an attempt to develop such a system does not appear feasible, even though it is being attempted by the AGC Aerophysics Department in their experiments using butane and water.

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In a porous (permeable) medium the mass flow rate is a function of the kind of gas employed, the characteristic pore diameter of the medium, the thickness of the porous medium and the difference in the squares of the pressures on either side of the porous medium. In the case of solid-propellant rockets utilizing aluminized propellants, an additional stipulation that the nozzle wall be held at a temperature above the freezing temperature of aluminum oxide is required. To maintain the nozzle wall at a constant temperature requires that the coolant mass-velocity be continually varied along the nozzle wall. This is best accomplished by maintaining a constant coolant reservoir pressure slightly greater than the combustion chamber pressure and either varying the nozzle wall thickness or varying the nozzle wall permeability in such a way as to accommodate the pressure and temperature gradients existing in the nozzle. Knowledge of the pressure and temperature distribution in the entrance, throat and exit sections of the nozzle is necessary for the design of the porous section.

Experimental investigations of the attainable capabilities of transpiration cooled systems have been carried out by a number of other agencies. For example, tests were performed on various porous materials specimens using an electrical resistance heating technique to determine whether water could be passed through the specimen, could be vaporized within the specimen, and could hold the material temperature below melting with high resistance heating rates. The specimen used was porous sintered stainless steel. It appeared that heat transfer rates up to 2,000 BTU/ft^2-sec could be accommodated this way. Since the steam issuing forth reduces the heat transfer rate from the boundary layer, this system might work for boundary layer heat transfer rates up to and exceeding 10,000 BTU/ft^2-sec.

As may be seen, delineation of $\dot{q}$ or Q^* values for any of the possible transpiration coolants would be a function of the quality of the porous medium through which the coolant transpires and the various operating parameters of the particular motor. Q^* for various candidate coolant materials ranges from 2,000 BTU/lb for air to 12,000 BTU/lb for hydrogen (H_2O - 2,500, He - 4,100, Li - 8,500) excluding the weight of associated inert parts.

If the coolant is a liquid or a gas it will have to be stored in a container outside the nozzle. The lightest container would be a spherical one, but a design based on a spherical container would occupy more space than one based on a cylindrical one. For the coolant weights given above, the actual container weight would probably be less than the weight of the nuts, bolts, and other attachments to the container. These might be about two pounds. The container should be at least 0.015 inch thick. A good material would be titanium because of its relatively low density, compared to steel. Connecting the container to chamber pressure and tapping the aft end for coolant delivery would provide an excellent pressure source. This would ensure a driving pressure always greater than the pressure at the throat. X650

The feasibility of transpiration cooling has been demonstrated by various investigators, but several problems still exist concerning its applicability to solid rocket nozzles. The problem areas (clogging, quality control of wall permeability, inert weight, variation in coolant requirements) must be thoroughly investigated before any firm conclusions can be reached about transpiration cooling. The significance of all the factors mentioned cannot be determined without an experimental program. Clogging, for example, may not occur. Some of the problems mentioned, such as mixing with the main gas stream and conduction in the boundary layer, are probably not decisive.

On the basis of this preliminary investigation the areas in which solutions might best be found experimentally are: flow stability, actual amounts of coolant necessary, plugging of pores by aluminum oxide, clogging by incomplete decomposition or combustion, and the effect of coolant injected tangentially to the wall.

(b) Film Cooling

Attempts to cool the nozzle area of solid rocket motors have been made utilizing a film of boiling liquid as the only cooling agent. An encompassing attempt was made on Project SQUID by Zucrow, Beighley and Knuth at Purdue University in 1950. They found that the system had application for overcoming local hot spots in the chamber area, but that the coolant film could not be uniformly maintained over large areas. If enough coolant was used to completely cool the entrance and throat areas of the nozzle, the quantities of coolant necessary were more than twice that expected because of the non-uniformity of the coolant film. These disadvantages were never completely overcome, although this type of cooling is often used for cooling small (40) critical areas which do not lend themselves to other more efficient cooling methods.

In the presence of aluminized propellant gases, it is necessary to keep the nozzle walls at sufficiently high temperature (of the order of $4,000^{\circ}$ F) to prevent the condensation of aluminum oxide. It is unlikely

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that a coolant material will soon be available which may be injected in an even film along the nozzle entrance section and maintain the wall at the required temperature. Thus, because of the restriction imposed on the boiling point of the coolant by aluminized propellant systems, it is unlikely that this type of cooling will find immediate usage.

The problems involved with a gas-film coolant are similar to those described in the propellant cooling section below, (the Prandtl-Karman mixing length is quite small). A distribution system which would diffuse the coolant gas and maintain an even boundary layer at a temperature above the limit imposed by aluminum oxide is quite difficult, but perhaps not impossible to achieve. Some experimental work has been done on this type of cooling (41) but a successful system for solid rockets has not as yet been developed. Film cooling may be used, however, to a limited extent, for cooling combustion chambers of liquid rocket motors.

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Along the lines of film cooling, consideration may be given to the hybrid sweat-film method of cooling in which advantage is taken of the highly efficient effusion (transpiration) cooling method in the region of highest heat flux (i.e., the region of the throat) and of the less efficient (also less complex) liquid film cooling method in the regions of lesser heat flux.

In order to employ this technique, the nozzle must be fabricated of both permeable and non-permeable materials. The region a short distance on either side of the plane of the throat would be constructed of a permeable material while the remainder of the nozzle would be constructed of non-permeable material.

The cooling technique involved is to force an excess of liquid through the highly permeable section of the nozzle and allow the excess liquid to flow over the non-permeable portions of the internal surfaces of the nozzle wall to provide them with liquid film cooling.

This technique has not been experimentally demonstrated; however, analytic solutions have been proposed (42). It is felt that this hybrid type cooling offers one of the highest thermal efficiencies per unit weight of cooling system but is also one of the more complex, less understood systems.

<u>(c) Evaporative Cooling</u>

The evaporative nozzle-cooling system shows great promise as a candidate cooling system. Because of the high coefficients of heat transfer in the nucleate boiling range and the mobility of a boiling material, it should be possible to maintain a large temperature difference across the boundary layer of the nozzle stream. The nozzle liner would have to be comparatively thin in order to keep the temperature drop across it low.

A coolant should be chosen which could add impulse to the system. Lithium, sodium and magnesium are among those considered.

Experiments were begun in late 1957 at Allegheny Ballistics Laboratory on evaporative cooling utilizing water, mercury, sodium, magnesium, and sodium magnesium alloys as coolants. Presumably that company has cooling systems which are highly developed. There are technical prob-
lems which must be overcome before such a system can be adapted for full X
scale use, but, because of the lightweight and basic simplicity of the 6
system, it bears a great deal of promise. The experimental work done by 5
Allegheny Ballistics Laboratories shows that the system may be developed 0
under a particular set of conditions even though the heat-transfer information on candidate coolants is fragmentary and the mechanics of boiling are not completely understood. The ABL experiments indicate that coolant weights of 0.05 percent to 0.1 percent of the propellant weight should be adequate for the system, depending upon the coolant material and the ratio of effective coolant to total coolant. The problem of maintaining a reservoir of coolant at the evaporation temperature is the most serious design problem which must be solved. Utilization of this system requires high rates of heat transfer which necessitate use of a liner material having a high coefficient of thermal conductivity. Tungsten and graphite are two such candidate materials.

The effect of a small percentage of lithium and other candidate coolant materials on stability of combustion of aluminized propellants should be investigated to determine whether or not they can be injected into the chamber without detrimental effects. There is a possibility that a candidate coolant could cause combustion instability if injected into the combustion chamber. Chamber injection should be considered, however, since the heat lost to the cooling system could be returned to the propulsion system with the possibility of adding impulse, especially if lithium can be used as the coolant.

The boiling temperature of the coolant can be controlled, to a certain extent, by controlling the pressure in the boiling-chamber through the location of the injection point in relation to the chamber-nozzle configuration. In general, higher heat transfer rates to boiling liquid metals are obtained when boiling occurs at higher pressures and the nucleate-boiling temperature range may be extended. This has been demonstrated experimentally (43). Of the coolants considered, lithium is the most promising.

The advantages of lithium are:

1. High heat of fusion and vaporization

2. Wide temperature range between melting and boiling

3. Boiling point is in the proper temperature range to allow a high heat-flux through the liner wall.

4. Low molecular weight

5. Solid at room temperature

6. Possible augmentation of impulse when injected into the chamber.

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These advantages are discussed in detail (44); however, the final choice of the coolant will depend upon the boiling temperature and the required temperature at the hot-side of the nozzle wall in order to maintain boiling in the nucleate range.

(d) Chemical Reaction Cooled Motors

Another phenomenon which can contribute to cooling, if used properly, is the heat-energy exchange taking place during chemical reactions. By inducing endothermic (heat absorbing) chemical reactions near or at the hot wall, a wall can be cooled. The escape of gases resulting from the reaction may also delay or prevent the deposition from aluminized propellants of aluminum oxide at the wall.

Two types of endothermic reactions are possible:

1. the dissociation reactions of exothermic compounds (heat addition is needed for their dissociation), and

2. the formation reactions of endothermic compounds (heat is absorbed during their formation).

The simplest case of chemical reaction cooling of a hot wall would be the decomposition or cracking of an exothermic compound impregnated in a porous wall. The heat transferred from the wall to the compound would raise the temperature of the compound sufficiently to cause its decomposition. As the decomposition products escape through the pores, they will absorb more heat from the wall. Furthermore, the decomposition gases, while entering the main gas stream, reduce the heat transfer coefficients by mass addition to the boundary layer. Should some of the decomposition products react with each other inside the wall to form new endothermic compounds, then the heat absorption capacity would be increased further.

The desired properties of high heat-absorption capacity chemical coolants are:

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1. High specific heat, so that large amount of energy is absorbed during heating of the coolant.

2. The changes in free energies of formation, ΔF, should favor the decomposition of coolants with the formation of endothermic compounds.

3. The coolant should consist of exothermic compounds absorbing heat during decomposition.

4. Low molecular weight of the products of decomposition and of subsequent endothermic reaction, produce large volumes of gases per pound of coolant. X650

5. High specific gravity, so that it is compact.

6. If possible, the coolant should burn upon entering the inside of the nozzle or chamber and thus contribute to the total thrust of the rocket.

7. No detrimental reactions between the wall and the coolant should take place. If possible, the coolant should act as an inhibitor of reactions between the wall and the propellant gases, thus preventing chemical erosion of the cooled wall.

In general, the lower the molecular weight of a gas, the higher will be its specific heat or the greater will be the amount of heat the gas will absorb during heating. From this point of view, compounds decomposing into hydrogen-rich gases are best heat-absorbing compounds. Hydrogen has the highest specific heat of all the candidate gases. It is followed by lithium hydride, (LiH), methane (CH_4), ammonia (NH_3) water (H_2O), etc.

So far three types of chemical reaction coolants have been found to possess a number of the desired properties listed above. They are the linear saturated hydrocarbons, nitrogen rich organic compounds, and miscellaneous inorganic salts with high heats of decomposition.

From a high specific heat point-of-view, paraffins $\left(C_nH_n{+2}\right)$ appear to be good heat-absorbing coolants mainly because of their high hydrogen content.

When considering the free energies of paraffins, one sees another good feature. Paraffins, like all other hydrocarbons, have a greater

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tendency to decompose at higher temperatures*, as indicated by their free energies. Also, the higher the molecular weight of the paraffins, the higher their free energies. This means that with increasing temperatures there is a tendency for higher paraffins to form lower ones, and for lower paraffins to form olefins, aromatics and acetylenes. Paraffins are exothermic compounds while olefins, aromatics and acetylene are endothermic compounds.

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The dissociation of paraffins and the formation of olefins, aromatics and acetylene will absorb more energy than the association of paraffins into carbon and hydrogen. Therefore, in any such design, the paraffins should be subjected to a very high heat flux for a very short duration of time. It should be noted that in some advanced uncooled solid motors, at wall temperatures of 3,500° F, about 33,000 Btu's/ft^2-min are transferred from hot propellant gases to the wall. To illustrate the decomposition of paraffins, assume that wax can be cracked into acetylene and hydrogen; then the gases are allowed to escape into the main gas stream at 3,500° F. The following chemical reactions may occur:

	Heat absorbed
Heat A + $C_{20}H_{42}(s) \longrightarrow C_{20}H_{42}\ (l)$	94 Btu/lb
Heat B + $C_{20}H_{42}(l) \longrightarrow C_{20}H_{42}\ (g)$	58 Btu/lb
Heat C + $C_{20}H_{42}(g) \longrightarrow 5C_4H_8\ (g) + H_2$	715 Btu/lb
Heat D + $5C_4H_8\ (g) \longrightarrow 10C_2H_2\ (g) + 10H_2$	3,400 Btu/lb
Heat + $C_{20}H_{42}\ (s) \longrightarrow 10C_2H_2\ (g) + 11H_2$	4,267 Btu/lb

Heating up of product gases to 3,500° F:

C_2H_2 - 2410 Btu/lb

H_2 - 12,900 Btu/lb

Heat absorbed: $\frac{260}{282} \times 2410 + \frac{22}{282} \times 12{,}900 = 3{,}225$ Btu/lb

Total heat absorbed 3,225 + 4,267 = 7,492 Btu/lb

*Acetylene has the distinction that while it is less stable than ethylene at low temperatures it becomes more stable at 2,100° F. At high temperatures it becomes more stable than any hydrocarbon, but it should be remembered that it is less stable than the free elements.

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If $C_{20}H_{42}$(s) were cracked into C + H_2 only, then approximately only 3320 Btu/lb would have been absorbed.

The coolant gases exiting from the wall and entering the boundary layer will tend to reduce the gas film heat transfer coefficient to the wall. Upon mixing with propellant gases, the coolant gases will burn at a flame temperature in the neighborhood of 6,000° F generating about 25,000 Btu/lb. Literature review indicates that the overall effect of gas injection with exothermic chemical reaction in the boundary layer will be the reduction of heat transferred from propellant gases to the wall. However, the increase in heat transferred to a nozzle by exothermic chemical reaction of injected gases in the boundary layer may counterbalance the desirable mass transfer cooling effects of the injected gases.

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The above example of the use of wax as coolant illustrates only a few possible dissociation and recombination reactions. More likely, besides acetylene and hydrogen, a number of olefins and short chain paraffins would be produced, which would reduce the heat absorbing capacity of the paraffin coolant.

There are a number of long-chain paraffin type compounds exhibiting very weak carbon to carbon links. An example would be polyisopropylene

$$\begin{array}{l}(\text{-CH-CH-})_n \\ \qquad\;\;\; | \\ \qquad\;\; CH_3\end{array}$$

Under very severe conditions this polymer will tend to decompose primarily into a monomer; then into methane, ethylene, acetylene, and finally carbon and hydrogen.

The breaking up of exothermic compounds, such as amines and amides, and the formation of endothermic compounds, such as cyanides, offers good possibilities for chemical reaction cooling. Nitrogen-rich polymers, such as melamine resins, urea resins, nylon, polyurethane, and others, could be impregnated into porous walls. At very high temperatures and high heat flux rates, the likelihood exists that some of their decomposition products may combine to form hydrocyanic acid.

Again, this process of decomposition and later of the formation of endothermic compounds, is comparable to the cracking of paraffins and the formation of acetylene, olefins, and aromatics. The heats of decomposition and of formation that are involved are of about the same magnitude as for paraffins.

A preliminary screening of inorganic compounds showed that ammonium salts and alkali hydrides have exceptionally high heats of decomposition.

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Also, their gaseous products of decomposition have high specific heats (an inherent property of light elements).

Perhaps the most impressive compound is lithium borohydride ($LiBH_4$). At one atmosphere pressure its decomposition starts at about 532° F. This endothermic compound absorbs 3,830 Btu/lb during decomposition. Heating $LiBH_4$ to 3,500° F will absorb about 8,500 Btu/lb. The compound contains about 18.4 percent by weight of hydrogen gas which should be very effective in mass addition to the boundary layer. The heat of combustion of $LiBH_4$ at 3,500° F is about 32,000 Btu/lb. Other lithium compounds appear also to be very effective coolants mainly because of their high heats of decomposition and high specific heats.

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Among the ammonium salts, ammonium chloride, ammonium carbonate, ammonium phosphate and ammonium perchlorate were investigated. Their heats of decomposition vary between about 500 Btu/lb and 1,428 Btu/lb.

While the analytical approach gives us an understanding of this method of cooling and the degree of cooling to expect, experimental data is still needed for the design of nozzles. Specifically, the heat-absorption capacities of coolants have to be determined experimentally, and the problems involved in impregnation of porous materials have to be solved.

(e) Propellant Cooled

The objective of this type system is to utilize low-flame-temperature solid propellant as a source of gas for cooling nozzles to temperatures at which they will neither erode nor melt. The solid propellants would be located in the converging section above the throat of the nozzle. During combustion the low-flame temperature propellants generate relatively cool gas which, when directed properly, will cool the nozzle by film or by transpiration cooling.

Conventional film cooling methods have the disadvantage that the coolants, coolant container, and a pumping mechanism add weight to the rocket motor. In using low flame temperature propellants as coolants we dispense with the pumping mechanism. And because the coolant propellants have potential chemical energy, the release of this energy adds thrust to the rocket motor.

Basically this approach is not new. Liquid rocket combustion chambers and nozzles are often cooled by regenerative and transpiration cooling techniques using liquid propellants as coolants. Upon entering the chamber or the nozzle, the preheated liquid propellants release during combustion both the heat energy they absorbed and their internal chemical energies. In low-flame temperature propellant cooling a similar effect is achieved. The coolant propellant releases its internal chemical energy during combustion and its combustion products generate a film along the wall of the nozzle, absorbing heat energy from high-flame temperature propellant gases that would otherwise be transferred to the wall and lost. Both energies are then converted to thrust in the nozzle. The high flame temperature propellants having a high specific impulse would provide the rocket's main propelling force. The low-flame-temperature propellants (usually with a slower burning rate and a lower specific impulse) can be used to cool the nozzle wall or other parts of the rocket motor.

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It is anticipated that the generation of a non-aluminized coolant gas film near the nozzle wall would also alleviate aluminum oxide deposition in the throat. Furthermore, it is believed that chemical erosion of the throat would be reduced or prevented if an inert or reducing coolant film were directed at the wall.

The critical regions of interest in the analysis of the low-flame-temperature propellant cooling are: (1) type and amount of low-flame-temperature propellant needed to cool the nozzle wall to the desired temperature, (2) the effect of addition of low-flame-temperature propellant on the thrust of the rocket motor, (3) the problems of mixing of high flame temperature propellant gases with low flame temperature coolant propellant gases, and (4) conceptual designs.

There are several groups of propellants which have a slow burning rate and a low flame temperature, and can be used for propellant cooling of nozzles. They are:

1. Non-aluminized propellants using ammonium perchlorate and hydrocarbon resins with flame temperatures of about 3,000° F and specific impulses of about 200 lbf-sec-lbm^{-1} at 1,000 psi chamber pressure,

2. Non-aluminized propellants using ammomium nitrate and cellulose acetate with flame temperatures of about 2,000° F and specific impulses of about 170 lbf-sec-lbm^{-1} at 1,000 psi chamber pressure, and

3. Non-aluminized propellants containing sodium azides or vinyl tetrazoles with flame temperatures of about 1,000° F. Specific impulse of these propellants is not known to the author at the present time.

A brief review of reports on heat transfer to the nozzle wall established that for engineering purposes at propellant combustion temperatures up to 5,500° F more than 90 percent of the heat transferred to the wall was by forced convection, the remainder being by radiation.

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At propellant combustion temperature (T_A) of 5,450° F, and chamber pressure of 1,000 psi, the range of heat transfer coefficients from hot propellant gases to the nozzle wall varies from 700 Btu/ft^2-hr at the end of the diverging section. As can be seen from the above data the throat area has to tolerate the highest heat flux rate.

For an uncooled nozzle of 1.25 ft^2 surface area, average heat transfer coefficient, hg, of 700 Btu/hr-ft^2 a propellant combustion temperature T_A, of 5,450° F and for a 60 second duration firing, the cooling requirements to keep a wall at 4,450° F are 14,600 Btu per rocket firing. For each 1,000° F decrease in the wall temperature the cooling requirements are essentially multiples of 14,600 Btu.

It is apparent from the above that the lower the temperature of the wall the greater the amount of heat transferred to the wall and the greater are the cooling requirements.

$$\frac{T_W - T_A}{T_B - T_A} = \frac{28Pr_A^{2/3}}{\frac{195(32)!}{(39)}} \; \frac{Re_B}{\frac{(7)\,!}{(39)}} \; \frac{0.2\frac{(\mu B)}{(\mu A)} \quad 0.2\frac{(C_{PB})}{(C_{PA})}}{(0.0288)} \left(\frac{X\rho_A \; V_A}{S\rho_B \; V_B}\right)^{0.8}$$

where

T	= temperature
Pr	= Prandtl number
Re	= Reynolds number
μ	= viscosity
C_P	= specific heat
X	= distance from leading edge of wall
S	= slot width
V	= gas velocity

and subscripts

A designates the hot gas
B designates the coolant gas
w designates the wall

This equation has been developed for turbulent flow assuming that instead of the gas being ejected through a slot for film cooling, a heat sink of the strength $C_P \; B^{SV_B} \left(T_B - T_A\right)$ is placed at the leading edge of the plate to be cooled. Although the Tribus-Klein equation has been developed for flat plate cooling, the results should give a good indication of film cooling under different flow conditions.

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For the rough order of magnitude determination of the amount of propellant required to cool the nozzle walls to desired temperature, it is felt that the use of the Tribus-Klein equation is justified. In this analysis it is assumed that the coolant gas is ejected through a slot located upstream of the nozzle throat area. Propellants B-1 (adiabatic flame temperature, $T_B = 1{,}850^\circ$ F, $I_{sp} = 174$ lbf-sec-lbm^{-1}) and B-2 (2,780° F, $I_{sp} = 197$ lbf-sec-lbm^{-1}) are used for nozzle cooling. It is also assumed that the solid rocket motor whose nozzle is being cooled contains 825 lbs of main propellant with a specific impulse of 259 lbf-sec-lbm^{-1}. The combustion temperature of this high-flame-temperature propellant is 5,450° F. The rocket firing duration is 60 seconds.

From the definitions of the specific impulse and thrust it can be shown that if part of the high-flame-temperature propellant is to be replaced by the low-flame temperature propellant, then the quantity of the low-flame temperature propellant required to keep the thrust the same is approximately equal to the ratio of the specific impulses times the quantity of the propellant removed. In equation form:

$$W_B = \frac{I_{SPA}}{I_{SPB}} \times W_A \text{ Removed.}$$

The movement of gases entering the nozzle depends to a large extent on the configuration of the converging section and on the propellant charge design. In some motors there is a definite "cork-screw-with-changing-pitch-type" gas movement. This movement is not the random molecular motion of gases which is responsible for heat transfer. Rather it is a turbulent motion of whole layers of gases which produces a rotating flame that will tend to destroy the cooling effects of the low-flame-temperature propellant gases. It is essential, therefore, that conceptual designs for propellant cooled nozzles prevent rapid destruction of the coolant film. This may be achieved by: (1) manufacturing slots in the low-flame-temperature propellant to assure vertical channelling of the main propellant gases; (2) creating a combustion zone for the low flame temperature gases above the converging section which would prevent rapid mixing of the hot and cool gases. The coolant gases would be sucked into the converging section by the main propellant gas stream causing film cooling of the nozzle.

(3) Testing Requirements

The urgency of military and civilian missile programs, and the rapid advances in propellant technology dictate that the inadequacies of our present theoretical methods be circumvented by experimental

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determinations of heat transfer parameters of the specific propellant-nozzle designs under development or study.

Such experimentation is generally carried on by static firings of many highly instrumented sub-scale motors and by subsequently confirming the data from these sub-scale tests in static firings of a small number of full scale motors. Unfortunately, even the heat transfer data obtained in sub-scale and full scale motor firings is questionable, and the final proof of satisfactory nozzle designs is based on repeated, reliable, successful performance of the design, in full scale firings. The heat transfer data obtained from successful firings frequently cover a sufficiently large range of values that correlation with data from unsuccessful firings, in which failure was attributed to breakdown of the thermal protection system, is difficult and dubious.

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The basic parameters measured in instrumented firings are gas pressures, gas temperatures, wall temperatures, and thrust, all versus time. These data together with other thermodynamic data (generally extrapolations) are then used to deduce gas velocities, Langmuir film thicknesses, viscosities, thermal conductivities, diffusivities, etc. which are used to determine the Nusselt, Prandtl, and Reynolds numbers present in convective heat transfer calculations.

With the possible exception of thrust data, the information gained in static firings contains inaccuracies stemming from the measuring techniques used, and sometimes lacks in precision due to the quality or type of instrumentation employed. For example, measurement of nozzle wall temperatures is sometimes attempted by inserting thermocouples in the nozzle structure near the inner surfaces of the nozzle and extrapolating the observed temperatures to wall temperatures. These extrapolations imply knowledge of thermal conductivities which have not been determined for many of the nozzle materials at the temperatures of operation of present motors. Optical determinations of nozzle temperatures are also restricted in usefulness due in part, at least, to the lack of total emissivity data on nozzle materials at very high temperatures as well as to the transparency factor of exhaust gases.

The very promising infra-red spectroscopic techniques (under development at the National Bureau of Standards) for measuring temperatures are limited by the lack of highly precise, sturdy instruments for use at the test stands, and by the need for further study of the clever manipulation of spectral "windows" and opacities to obtain temperature profiles of the entire nozzle wall and propellant stream. This latter technique may also prove extremely useful in determining the gaseous species and intermediate chemical reactions and products within the stream and provide information as to the reaction energies (thermal) and the locations in which they occur.

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Accurate determination of $\dot{q}$ and Q^* for any thermal protection system will depend on: (1) thorough investigation of the thermal and physical properties versus temperature of the nozzle materials used, (2) improvement of the measuring techniques and instruments used in static firings, (3) determination of the dynamic composition of the propellant gases, (4) determination of the dynamic properties and composition of the thermal protection systems, particularly where ablation, chemical reaction, or phase changes take place, (5) development of more accurate correlations of present gas dynamical parameters (Nu, Pr and Re numbers), (6) development of new functions and parameters which more rigorously describe convective heat-transfer phenomena.

With respect to the areas of investigation enumerated above, it is envisioned that prospective research and development programs would include:

(a) Thermal Properties of Materials Versus Temperature

1. Specific Heats (of all proposed materials in all pertinent phases)

2. Thermal Conductivities

    a. Inert Nozzle Materials
    b. Coolant Materials

        1. In all pertinent phases

3. Viscosities of Liquid Coolants

4. Total and Spectral Emissivities

    a. Infra-red spectrum
    b. Visual spectrum

5. Thermal Expansion

6. Vapor Pressures and Energies of Formation, Fusion, Dissociation, Etc. where such information is lacking and needed.

(b) Improvement of Measuring Techniques and Instrumentation

1. Temperature

    a. Development of infra-red spectrographic techniques and any other methods which do not alter or affect the temperature profile by their presence.

(c) Determination of Dynamic Gas Composition

1. Infra-red spectroscopy

2. Visual spectrum spectroscopy

(d) Determination of Dynamic Properties and Composition of the Various Types of Thermal Protection Systems

1. Erosion rates

2. Ablation rates

3. Intermediate Compound formation

4. Film Cooling effects

5. Items listed in (a) above versus time

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It is felt that immediate and intensive programs to improve experimental techniques and develop better theoretical understanding of nozzle heat transfer would be of great assistance in the design of successful, reliable cooled nozzles.

d. Recommendations

Development of thermal protection systems for solid propellant rocket motor nozzles must be carried on from the standpoint of investigating each of the candidate types of systems, as they apply to the specifications of particular motors. Accurate determination of heat transfer characteristics of each type of TPS must, of necessity, precede or be concurrent with fabrication and testing of the system. The types of data required are outlined in c.(3) above. The selection of specific materials for the various TPS is entirely dependent upon the system involved and will range from refractory materials for flame barrier and heat sink components to low melting reactive metals for liquid metal coolants, to organic materials for chemically reactive or ablative coolants.

Although fabrication development of the selected materials and configurations may require intensive effort, selection of TPS's and their materials should be based primarily on their thermal protection capability.

5. RADIATION-COOLED ROCKET MOTORS

a. Description of Components

Many space vehicles will utilize liquid chemical propellants and it is likely that attitude control, mid-course correction thrust, etc., will be accomplished with radiation-cooled, bi-propellant rockets. Normally, this system would utilize the same propellants as the main propulsion system (for solid propellants, a self-contained system would be required) and would operate at a low pressure to minimize system weight. Propellant valves attached to a low manifold volume injector will be actuated electrically with a very small actuation delay.

b. Operation Characteristics

Control rockets will operate on a demand cycle that can occur as pulsed bursts or as a continuous firing for the required duration. By using the "pulse rocket" concept, temperatures of the radiation-cooled thrust chambers are reduced; however, the thrust chamber must endure many pressure cycles.

Since the cooling of the rocket will be by radiation only (heat-sink cooling cannot be used because of cumulatively long duty,) the chamber pressure will be low. In practice, this pressure will probably be fixed by the maximum permissible skin temperature of the thrust chamber. The chamber pressure for this type of motor will vary between 5 and 50 psia depending on the propellant combination, mixture ratio, and operating cycle. Pressure transients will be of the order of 10 milliseconds for accurate control of total impulse. Thrust-chamber wall thicknesses will vary from 5 to 100 mils depending on size.

c. Environments

The complete system will be exposed to the vacuum of space so that vapor pressure of the chamber material places one upper limit on the material. The temperature of the chamber material will be determined by the local heat transfer conditions from the exhaust gases, the material emissivity, and the shape factor for radiant interchange. The maximum temperature will normally occur in the throat region and will be between 2,900° and 3,500° R. At these temperatures the reactivity of the chamber material with the exhaust gases can become important and must be evaluated for each condition.

The exhaust gases are normally stoichiometrically reducing; however, under actual firing conditions, localized, oxidizer-rich streaking exists due to improper propellant mixing or stratification. In the case of the pulse-rocket design, the high vapor pressure of the normally-employed

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oxidizers (compared to fuels) result in the interior thrust-chamber walls being subjected to oxidizer vapor during temperature-decay periods which occur between pulses. (Pulse-fire rates may vary from 10 to 100 times per second.) The interior of the thrust chamber is therefore subjected to simultaneous corrosive action of hot reducing and oxidizing gases, either of which can bring about conditions leading to erosion or embrittlement. In the latter case, embrittlement results from the formation of interstitial compounds such as carbides, nitrides, hydrides, and oxides. The use of coatings which are stable under these conditions is, therefore, a prime necessity, particularly when refractory metals such as molybdenum are used as the radiation-cooled thrust chamber material. In addition, oxidation resistance is extremely desirable for thrust chamber exteriors, since it is advantageous in the early phases of development to perform these initial firings when exposed to air atmosphere, and avoid the use of costly vacuum-firing test conditions necessary for outer-space simulations.

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Due to the many pressure and thermal cycles the system will experience, the cyclic life of the material will be extremely important, as will the resistance to thermal shock if a nonmetallic construction material is used. Because of this cycling, the use of short-time tensile strength data for design purposes is questionable and needs to be verified by multiple cycle and stress rupture tests on the more promising materials.

d. Potential TPS

(1) General

The most promising thrust chamber materials are the refractory metal alloys of molybdenum, tungsten, tantalum, niobium, and titanium (although some interest in pyrolytic graphite and graphite coated with silicon carbide and titanium nitride has been evidenced). Since radiation is the primary means of cooling, any improvement in radiative properties of the material must come through use of coatings. In the use of coatings, their stability under relative high vacuum and thermal conditions require evaluation and development to maintain chemical stability in spite of diffusion and evaporation effects.

The high thermal conductivity of most of the refractory metals allows conduction to be used to reduce the severity of the throat conditions without introduction of weight penalties. The use of an internal coating as a thermal resistance does not appear to be practical, because of the low heat fluxes involved in this type of system.

(2) Required Material

The materials required for application to radiation-cooled rocket motors should exhibit modest strength and consistent properties at high-cycle operating temperature, high emissivity, resistance to both reducing and oxidizing atmospheres, thermal shock-resistant characteristics, and a very low vapor pressure as well as fabrication characteristics consistent with thrust-chamber design considerations.

(3) Testing

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The physical properties of potential materials need to be evaluated in the operating temperature range with major emphasis on emissivities, creep, and thermal fatigue strength of the materials. These properties then serve to screen candidate materials for model testing as actual subscale, radiation-cooled rocket motors in altitude chambers. The use of super alloys for radiation-cooled thrust chambers is prohibited because of their low melting point and loss of strength at temperatures at which incipient melting occurs. Titanium alloys can be considered for this application because of their higher melting point (compared to super alloys) and retention of useful strengths at short exposure times (10 sec. duration) in a portion of the temperature range of interest. The low strength of titanium alloys as compared to molybdenum in the low portion of the operating temperature range, may be more than offset in a given application by its higher emissivity and oxidation resistance.

(4) Recommendations

Material programs to determine the pertinent high temperature properties of potential materials are a necessity. Research on coatings to improve the emissivity and stability of these materials and coatings which will not degrade under thermal and pressure cycling and under high-vacuum conditions are required.

6. ROCKET CASINGS

Successful operation of rocket motors requires suitable systems or devices for thermal protection of the rocket casing.

Within the rocket motor, combustion of the propellant releases large quantities of thermal energy; consequently, the interior of the case may be exposed to temperatures of several thousand degrees Fahrenheit during the firing operation of the motor.

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Aerodynamic heating on external surfaces of the motor casing is somewhat less severe; nevertheless, considerably elevated temperatures may result.

A third source of elevated-temperature exposure results from "wash back" of the hot combustion gases near the base of the rocket motor casing. Jetevator action and/or aerodynamic aspiration of the hot gases exiting from the nozzle, can cause localized overheating on the outside of the vehicle, particularly at the aft head.

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In all three cases adequate thermal protection to the rocket motor casing is necessary in order to maintain its temperature well below the maximum value at which it can safely carry structural loadings as required.

The success of solid-propellant rocket motors is largely due to their simplicity of design and concomitant reliability. The rocket designer is faced with three possible choices: (1) the use of extremely thick and heavy hardware (heat-sink designs), (2) the use of cooling systems, or (3) the use of suitable thermal protection (insulation) systems. Optimum design generally requires that the motor be as lightweight as possible and be operated without cooling devices, e.g., pumps, motors, valves, coolant supply, etc. The use of thermal protection systems has invariably proven the most appropriate design philosophy.

The protection of the outer, load-carrying structure from aerodynamic heating, is complicated by the severity of heating environments through which the vehicle must survive, aerodynamic smoothness, and, in many cases, envelope dimensional requirements. This problem, however, is not peculiar to rocket motors. Aerodynamic heating has been a consideration in the design of missile nose cones, leading edges, and other re-entry bodies, and high-speed aircraft. In general, aerodynamic heating is considerably less severe for rocket motor casings. Protection against "wash-back" heating may generally be considered similar but less critical than that for internal heating. Accordingly, the present discussion need not concern itself with these problems, and will be restricted to the more critical problem of internal thermal protection as is peculiar to rocket motors.

a. Definition of the Environment

(1) Description of Components

For purposes of internal thermal protection, the solid-propellant motor casing may be considered as comprising three distinct sections: (1) forward head (forward closure), (2) cylindrical chamber, and (3) aft head (aft closure).

b. Thermal Requirements

The thermal environments are most severe at the aft head where the insulation material is exposed for essentially the entire firing period. Durations may range from a fraction of a second to minutes, depending upon the motor design. Chamber pressure may also range from about 100 to over 3,000 psi for various motors. The exact conditions will vary somewhat depending upon the solid-propellant grain design, being most severe for end-burning grains, and considerably less so for internal-burning grains.

During operation of the rocket motor, the thermal protection system X
at the aft end of the motor casing is exposed to moderately high erosive 6
forces and heat fluxes. Combustion gas temperatures range from about 5
3,000 to over 6,000° F; and gas velocity is about Mach 0.1 - 0.5, 0
increasing toward the nozzle. In addition, small scale turbulent gas flow often occurs causing localized erosion, particularly as the gas velocity increases rapidly as the combustion gases approach the nozzle entrance section.

Requirements for thermal protection of the forward head and the chamber section are considerably less severe. Duration of thermal exposure is considerably shorter than for the aft head section. For internal-burning grains, exposure may occur only at the last fraction of a second or so of the firing operation. For end-burning grains, exposure time will decrease toward the forward head of the motor. In all cases the heat flux to the case insulation forward of the aft head, is relatively moderate compared with that at the aft head of the motor casing. Combustion gas flow in the cylindrical chamber section is essentially parallel to the surface of the insulation and at a relatively low velocity (generally less than 100 ft/sec). Consequently, erosion is not a critical factor in the cylindrical and forward end sections.

Complicating factors are the chemical reactivity of the extremely high-temperature propellant combustion products with the thermal protection system, and possible mechanical erosive forces due to solid particulate materials in the propellant.

The extremely high temperatures result in depolymerization, melting, vaporization, and/or pyrolysis of insulation materials. The rates at which these processes occur generally increase markedly with temperature, resulting in reduced effectiveness of the insulation material.

Pressurization of the rocket case imparts mechanical stresses to the thermal insulation. In addition, the higher the chamber pressure and combustion gas density, the greater the degree of restriction for transpiration of gaseous products from organic thermal protection systems; such transpiration can effect a degree of cooling of the insulation system. On the other hand, at very low pressures many insulation materials tend to "blow," i.e., expand, due to entrapped gaseous decomposition products from the insulation material. Under such conditions, laminar-type constructions tend to delaminate and thus fail mechanically. Such "blowing" or delamination is less prone to occur at the higher chamber pressures; and pyrolysis of organic insulation materials tends to provide a harder layer of char. It is noteworthy that the formation of a hard, strong carbonaceous layer of char is extremely desirable. The char layer should provide an exposure surface which can resist thermal and mechanical ablation, and chemical attack. The char layer, being porous, permits passage of gaseous decomposition products from the insulation material below it, thus resulting in effective transpiration cooling of the thermal protection system and heat blockage at the boundary layer. On the other hand, at the higher chamber pressures a higher rate of heat conduction occurs, because of restricted transpiration of gases and greater combustion gas density.

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Flow of combustion gases over the surface of the insulation material induces a shear stress at the surface of the insulation. The higher the stress, the greater the degree of surface erosion, and the more likely the occurrence of "hot spots" due to localized turbulent flow.

In addition to the conditions described above, other factors determine the selection and design of an internal thermal protection system for rocket motor casings. In particular, it is essential that mechanical stresses and concomitant strains in the insulation system, do not cause failure thereof due to tearing, cracking, etc. In addition, the insulation system must be adequately bonded both to the casing and the propellant grain. Shrinkage during cure of the insulation and propellant should be kept as small as possible in order to minimize stresses within the insulation (and propellant) and at interfaces. Due to differential thermal expansion, temperature cycling effects may be critical in some instances. Other factors to consider are inertia and flight loadings, the attitude of the rocket motor during various stages, i.e., storage, handling, launching, and flight, and induced vibrations.

c. Potential Thermal Protection Systems

(1) Description of Various Solutions

Systems currently employed for thermal protection of internal surfaces of rocket motor casings, are generally largely organic in nature, and may be broadly categorized as two types: (1) flexible (elastomeric) systems, and (2) rigid systems. Elastomeric grain restrictors are often quite adequate for thermal protection in the cylindrical chamber section. Elastomers which have been found suitable include polyurethanes, butyl, neoprene, and Buna-N. Reinforcing fillers are frequently employed as compounding ingredients; these include materials such as silica powders and asbestos fibers. To a limited extent, graphite fabric is being employed, primarily as a thermal erosion shield for protection of the aft head.

Rigid insulation systems have found their greatest usage in aft and forward head insulations. Heat-resistant phenolic and epoxy resin binders are most commonly employed in conjunction with fabric reinforcements of asbestos, glass, and, in some instances, Refrasil. The epoxy resin systems offer superior fabrication qualities; however, to date, they are not as heat-resistant as the phenolic binders.

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(2) Required Materials and Properties

Properties of interest in the selection and design of internal thermal protection systems, include thermal conductivity, specific heat, and density over the entire operating temperature range (or, at least, to the decomposition point). To a lesser extent, mechanical properties over the operating temperature range are also important, particularly with rigid insulation materials such as reinforced phenolics. For refinement of insulation designs, information is desirable on heats of ablation under the thermal fluxes encountered at various stations within the rocket motor case, rates of decomposition, nature and quantity of volatile products produced, and char layer properties.

(3) Testing Requirements

Various devices have been employed for testing of insulation materials. For screening purposes, i.e., rating by comparative means of various materials and processing parameters, torch tests, especially oxyacetylene, are being widely employed. With suitable instrumentation, it is possible to obtain information such as the effective heat of ablation (decomposition) which includes all effects including melting, vaporization, etc., rate of dimensional change, rate of mass loss, and "back-surface temperature," i.e., temperature change as a function of time at a given distance from the exposed surface under a given heat input.

Subscale tests employing small rocket motors fired with the actual propellants and employing the insulation systems under test, might give more direct information; however, such tests are fairly expensive - considerably more so than the torch test. Additionally, exactly reproducible firings are difficult to achieve. Subscale motor tests, however, do have the advantage of providing the exact chemical composition of the propellant combustion products and more closely simulating the thermal and mechanical environments. It is generally conceded at the

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present time that such subscale tests do not permit direct scaleup to specific actual rocket motors. However, such subscale tests do permit more accurate comparative evaluation of candidate materials and fabrication parameters. Designs based upon such information generally are made redundant, i.e., overdesigned.

d. Recommendations

The thermal protection system adds considerable weight to the rocket motor, and is completely inert; i.e., it does not contribute positively to the propulsive thrust. Additionally, the volume occupied subtracts from the propellant capacity of the motor, thus reducing its mass fraction. Accordingly, it is desirable to employ the minimum thickness and weight of protective material. Propellant grain design can help to reduce thickness requirements by minimizing the time of exposure to the hot combustion gases. However, this surely can be no panacea for there are limits to compromises in grain design and points of diminishing returns in terms of the over-all efficiency and producibility of the rocket motor. It is highly desirable, then, to utilize a thermal protective system which has the maximum efficiency — i.e., minimum density and minimum required thickness.

To date, it has been, in large measure, possible to select and design internal thermal protection systems on essentially an empirical basis. Candidate materials and constructions are first selected by comparison of values for properties such as thermal conductivity and density, and comparative performance evaluation using torch tests and simulated or subscale firings. Selected materials and constructions are then introduced into the actual rocket motor being developed, and subjected to static firings. The condition of the protective material and construction is examined after firing; and decisions are made regarding modifications to the thickness and/or construction details, e.g., bond to propellant, bond to casing, addition of erosion-resistant thermal shield, etc. Motor firings are then conducted using the modified protective systems until satisfactory performance is obtained, within the limits of schedules and/or funds. Lack of adequate design criteria and properties' information generally precludes extensive application of more sophisticated approaches to selection and design of the thermal protection system.

It is readily apparent that there is considerable opportunity for elevating the current state of the art for internal thermal protective systems of rocket motor casings.

Development of improved materials is being pursued by various agencies and materials suppliers. Objectives include development of materials with lower heat transfer rates and lower densities, development of sacrificial ablative systems, and development of more

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erosion-resistant, nonablating systems (for heat shields, especially in aft heads). The ablation concept involves the orderly removal of material from the exposed surface, or just below the surface. This results in a transpiration-cooling effect and heat-blockage at the boundary layer, thus providing means for increasing thermal protection effectiveness. Such developments will become even more critical as the size of rocket motors increase, and as new, higher-temperature propellants come into use.

More emphasis is required in establishing efficient fabrication
processes for application and installation of thermal protection systems. X
In particular, fabrication techniques should be simplified to permit 6
reproducibility using semiskilled labor. Indeed, fabrication qualities 5
ought to be considered in development of materials. 0

Further work is required to establish more sophisticated design criteria. Gas dynamics and heat transfer studies and analyses, coupled with determination of applicable properties of thermal protection systems, can provide more quantitative criteria. Mathematical expressions thus derived should be correlated with performance results.

Inspection criteria and techniques also require further development. In particular, nondestructive methods need to be established for ascertaining the structural integrity of the thermal protection system, bond strengths, and dimensional tolerances.

7. POWER PLANTS FOR SPACE VEHICLES

a. Nuclear Engine

(1) Description of Components

Basically, the nuclear engine differs from the chemical rocket engine in the fact that the energy used to propel the system is accomplished not by the reaction of the propellants, but rather by the heating of a fluid by passing through a nuclear reactor and then expanding this heated fluid, in this case hydrogen, through a nozzle. In general, the nuclear engine consists of a propellant tank, a turbopump, valves, lines, thrust structure, reactor and nozzle.

Reactor

As presently conceived, the nuclear reactors for the next few generations of nuclear engines will be of the graphite solid heat exchanger type. This material, the hydrogen propellant, is heated while flowing through voids within the core. Even without a major breakthrough in technology, this concept will be capable of giving maximum system I_s of about 900 sec when graphite is utilized as the core material. The major effort for propulsion reactors in this country is presently being directed toward such graphite core reactors. A great deal of development effort will be required on other concepts such as the metallic or carbide-fast reactor, the liquid-core reactor, the cavity reactor, etc., before their technology is sufficiently advanced for application to a useful nuclear rocket vehicle.

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The power density of the first graphite reactors to be used in a flight engine may be of the order of 50 mw/ft^3 of core volume. Considering only normal development effort with no major breakthrough, the reactor for a nuclear booster engine may have power density levels greater than 200 mw/ft^3 of core volume.

Feed System

The hydrogen flowing from the tank goes through a booster pump and then into an axial-flow pump where the pressure is raised to 1,600 psi. Hot hydrogen gas downstream of the reactor core is mixed with cold gas coming from the pump, thus producing gas at 1,350° F to drive the turbine. The turbine exhaust may be fed to small nozzles to produce additional thrust, used for roll control or for thrust vector control of an upper-stage vehicle.

The conventional pump in chemical rockets has been the centrifugal pump. Since the ultimate objective of the nuclear propulsion system is a high thrust engine, the design of a pump with high propellant flows and very high heads have made it necessary to re-examine the pump design concept. It was determined that this pump should be the axial-flow type. Some of the advantages are lighter weight, better efficiency, and smaller envelope. Experimental results have indicated that efficiencies of the order of 85 percent are obtainable.

The following are the design parameters of the turbine:

1. inlet temperature, F - 1350
2. inlet pressure, psia - 750
3. exit pressure, psi - 69

The turbine was designed to operate on the combustion products of hydrogen peroxide, but, at a slightly lower efficiency, it can operate satisfactorily on pure hydrogen.

Booster Pump

To reduce the hydrogen tank pressure and without sacrificing the efficiency of the pump, a booster pump is required. A number of concepts for the booster pump are presently undergoing analysis.

The booster pump can be driven by a turbine using hydrogen in either liquid or gaseous form. A liquid-driven turbine creates fewer problems of thermal isolation than does a gas turbine. Because of the higher density of the liquid it is less bulky than the gas turbine. Liquid would be tapped off from the main pump, constituting an increased external pumping load. X650

A gear-driven boost pump can take power off the main turbopump. The power recovery factor of a gear system is practically 100 percent, in contrast to a turbine efficiency of about 70 percent.

A gear arrangement imposes greater restraints on component location.

A few of the many improvements that can be made in the feed system are:

1. Higher turbine operating temperature due to the use of high-temperature metals such as René-41, tungsten alloys, etc.

2. An efficient pumping system capable of pumping a boiling liquid. Thus, the nuclear radiation to the hydrogen propellant would go into heat of vaporization giving a small boiloff rather than into temperature rise which necessitates an increased tank pressure for a given net positive suction head. This would also allow the tank separation distance from the reactor core to be decreased.

Nozzle

The expansion nozzle is the conventional bell-type. For an upper-stage engine, the area ratio will be about 40:1 while for ground-launching the optimum area ratio is about 14:1. The heat fluxes in the nozzle are about an order of magnitude above those for chemical engines even when the nuclear engine is operating at 4,000° F. For example, a 1,500 mw reactor with 4,000° F gas at 1,000 psi P_c and a gas side-wall temperature of 1,500° F, will have a maximum heat flux of 36 Btu/sec in.^{2} when uncoated. The use of high coolant velocities and coatings will allow the nozzle to be adequately cooled when fabricated of conventional materials (e.g., Inconel-X). A possible coating for the nozzle is as follows: A 2 mil undercoat thickness of Type V nichrome followed by a 4 mil continuous graduation of zirconia and finally a 3 mil metallic carbide coating to protect the zirconia from reduction at high temperatures.

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As gas temperatures are increased, the heat fluxes will be further increased and use of such high-temperature tube materials as molybdenum and later tungsten will allow for adequately cooled nozzles.

Thrust Mounting

The tubular thrust structure is integrated with the propellant discharge ducting. The high-pressure hydrogen from the pump flows through the tubular thrust structure to the reactor dome.

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Analysis has shown that the lightest structure would be a variable thickness conical shell. Pressurization would be required for this structure. It is felt, however, that fabrication, assembly, and cooling problems delegate this type of structure to a future generation engine design.

A two-stage aluminum, tubular thrust structure is used in the present design. Since cooling of the thrust structure is required some distance away from the reactor core, the two-stage structure was chosen to allow the lower portion to carry the coolant. The lower portion of the structure then assumes the dual role of mount structure and high-pressure propellant ducting. The tubular-type thrust structure also offers the maximum flexibility for access to components and for possible variations in coolant routing. The upper thrust structure may, if desired, be utilized as the container for engine lubricants.

It is possible that further study of the upper portion of thrust structure may result in a stiffened, semi-monocoque structure or a conical shell structure. This final configuration will depend to a great extent upon the type of attach structure to be obtained from more detailed tank studies.

(2) <u>Operating Characteristics</u>

The nuclear rocket would be used as the upper stage of a satellite or space vehicle. The operating time is estimated at from 400 to 600 seconds. However the system must have restart capability.

(3) <u>Resulting Thermal Environment</u>

In this system the liquid hydrogen from the propellant tank travels first to a booster pump then to the axial flow pump from which it is discharged at 1,600 psia. From the pump exit, the propellant is transferred to the regeneratively cooled nozzle by the tubular thrust-structure. After cooling the nozzle, the hydrogen cools the reactor reflector and goes through the reactor core where it is heated to high temperature before expulsion through the nozzle. Current reactor state-of-the-art will limit this temperature to approximately 4,000° F. A small amount

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of hot gas is bled off downstream of the reactor and mixed with cold hydrogen to operate the turbine at about 1,350° F. The separation distance between the reactor core and propellant tank is about 20 feet. For this condition, no propellant shielding is required and the temperature of the hydrogen in the tank will increase at most only one to two degrees Fahrenheit due to nuclear radiation.

(4) Secondary Environments

Radiation from the power reactor in a nuclear propulsion system will
affect many areas of design in a flight vehicle. Heating effects and X
radiation damage must be evaluated with regard to their importance in 6
determining the design of components, structure and shielding. Cosmic 5
radiations constitute a second source of radiation, but its different 0
nature requires that it be analyzed separately. Both reactor and cosmic radiations are of importance in their effects on space vehicles, but neither presents problems that cannot be solved with existing design techniques.

(5) Potential TPS

(a) Discussion of Various Solutions

Tank Structure

For a given space mission there are three distinct and varied phases for which heat protection must be provided. These are the ground hold period, the boost phase, and coast phase of the flight in space. Because the fuel tank insulation is primarily designed by flight performance criteria, the fuel evaporative losses on the ground during a long hold-period with topping may become a significant cost item. If long hold periods with a filled tank are anticipated it may be economical to provide an outer insulative blanket on the tank side-walls which would be remotely removed immediately prior to liftoff.

Boost phase conditions which prevail are high temperature due to aerodynamic heating and large aerodynamic forces. This phase, although of short duration, dictates the structural requirements of the insulation.

During the coast phase of flight the only source of thermal energy is radiation from the sun and planets. By the use of radiation shields surrounding the tank, the heat flux across the tank wall can be effectively controlled. The properties of materials used in a solar shield are very important. Of prime importance are the absorptivity and emissivity of the surface when subjected to various types of radiation and body temperatures. Magnesium oxide and silver are two materials that show promise for use in a solar shield. These materials could be applied as coatings to a lighter base material such as aluminum. The number of shields required depends on the duration of coast flight and optimization of shield weight versus hydrogen boil-off weight.

There are many design options in selecting the heat protection technique. The insulation can be located internal or external to the tank wall, it can be integral or disposable during boost, or it can be bonded in place or mechanically retained.

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Internal insulation possibly offers a more continuous covering of the tank and protection against handling damage. However, undesirable features are apparent, such as difficulties in installation, in locating and repairing any leaks, as well as failures induced by thermal loading (differential expansion).

Two alternate external side-wall insulation systems require consideration. The aerodynamic shield portion of the insulation can be released in flight, leaving only the radiation shielding; or the entire assembly can be retained as an integral part of the tank. The disposable insulation has the immediate advantage of reducing upper stage weight. However, this system does have a major drawback in that it necessitates a system for attachment and jettisoning the blanket.

A lightweight radiation-type insulation system, such as Linde SI-4 and SI-12, has advantages. At present, insulation of this type is being used in some of the double-wall vacuum dewars for long-time storage of liquid hydrogen. Because this insulation is laminated from many thin layers of aluminum foil and fiberglass, it is an ideal radiation insulation. To be applied to the stage, it is laid on in continuous circumferential sheets and secured to the tank. This insulation method probably provides the highest thermal resistance with the lowest weight of any insulation material currently under consideration.

Reactor

The heating effects of intense radiation are of primary importance in the vicinity of the reactor. Pressure shells, control actuators, and thrust structure must be designed with some provision for cooling if they are to operate over the full duration of the reactor power cycle. Instrumentation must contend with effects of ionization and disruption of molecular lattice structure as well as the internal heating of the radiation flux.

Feed System

While heating effects are less important as locations are examined farther from the reactor core, they are still of importance with regard to sensing devices and control mechanisms. Certain radiation sensitive components may require shielding, such as gaskets, O-rings, Teflon bushings, and some kinds of electrical insulation.

Thrust Chamber

See description of coating for nozzle under section - a. (1).

(b) Recommended TPS's

Insulation of the fuel tank, some of the other components, and part of the thrust structure in addition to the coating of the nozzle is recommended. In addition it is necessary that the thrust structure, feed system and nozzle is protected by conduction i.e., in this case cooling by the flow of liquid hydrogen.

(c) Required Materials and Properties

The properties of materials used in TPS for this engine vary. Insulation located in close proximity to the reactor must be able to withstand nuclear radiation. The coatings mentioned in section - a. (1) for protection of the nozzle must be able to withstand high gas erosion.

(d) Testing Requirements

The coatings for the nozzle can be tested in static tests utilizing either the nuclear engine or a liquid oxygen-hydrogen chemical rocket engine.

The insulation for the tank system can be adequately tested in ground tests.

(6) Recommendations

In summary there are two types of thermal protection systems to be used in the nuclear engine. The nozzle, the tank and some of the components will utilize insulation. The nozzle, the thrust structure and the feed system will also use conduction as a thermal protection system.

b. Ion Engine

(1) Description of Components

Various power and ionization systems have been suggested for an ion rocket. The power source may be a nuclear reactor or the sun. The power conversion system for the production of the necessary electrical power may utilize a turbo-generator system, beta decay batteries, thermopiles, or solar batteries. Two promising techniques for the ionization of atoms employ surface contact or arc sources.

At present, the system showing the greatest promise for the necessary high capacity (in the megawatt range), very reliable, compact, lightweight power supply is the nuclear-turboelectric system utilizing a single-loop boiling alkali metal Rankine thermodynamic cycle. The alkali metal may be potassium, rubidium, sodium, or, possibly, lithium.

Any system intended for operation in space has the necessary limitation of rejecting heat by thermal radiation. The required radiator area to maintain the highest possible performance is inversely proportional to the fourth power of the effective temperature of radiation. This factor and the fact that the radiator will be the heaviest component of the power supply system indicates the need for a system capable of operating at high temperatures. A system utilizing an alkali metal as the thermodynamic fluid can meet this need. The components of such a system will include the reactor, a liquid-vapor separator, turbine, alternator, direct condensing radiator, and liquid pump. The reactor would probably be a fast reactor; the turbine, a multi-stage, axial flow, reaction type; the alternator, an electromagnetic inductor; and the pump or pumps axial inducer, mixed flow impeller types. The radiator would probably be constructed of tubes, of a material such as stainless steel, with cooling fins and a meteorite shield. The separator would be a centrifugal separator.

The propellant for the ion rocket would be an easily ionizable material, such as cesium. The propellant feed system components would include the supply tank, various valves and regulators to control the feed rate to the ionizer, a pump or other means of obtaining the necessary flow rate, a vaporizing chamber, and the interconnecting plumbing.

The surface contact ionizer appears most suitable for an ion rocket because of its simplicity, high ionization efficiency, and capability for long unattended operation. In applying the surface-contact source principle to ion propulsion it is desirable to flow the propellant through a porous ionizer so that it is ionized as it evaporates from the surface. Suitable porous ionizer materials, which must have high work functions, would include tungsten, tantalum, molybdenum, titanium carbide, etc.

High voltage electrodes placed aft of the ionizer provide the acceleration gap through which the ions are accelerated to high velocities.

(2) Operational Characteristics

(a) Power System

The turbine inlet temperature would depend on the alkali metal chosen for the working fluid and on turbine material limitations. From a thermodynamic standpoint, rubidium is promising in the temperature range from 1,600° F to 2,000° F, potassium from 1,800° F to 2,200° F, and sodium from 2,000° F to 2,600° F. Because of material limitations, it might be well to consider the maximum feasible turbine inlet temperature to be 2,200° F, particularly considering the long service life needed. The maximum reactor fuel temperature would be somewhat higher, less than 200° F higher, than the turbine inlet temperature. For minimal radiator area, the condensing temperature (absolute) should be about 3/4 of the absolute turbine-inlet temperatures. Thus, if we take 1,600° F to 2,200° F as the possible range of turbine-inlet temperatures, the corresponding range of temperatures for the radiator would be about 1,100° F to 1,500° F. In order to provide the necessary net positive suction head for the pump, the liquid must be subcooled below the condensing temperature. By further subcooling, the condensate may be used as a coolant steam for such things as bearings and alternator coils. The temperature, and therefore the size, of the radiator may thus depend on developments in such areas as high temperature alternators and bearings. The radiator will have to be made sufficiently strong or provided with armor to prevent penetration by meteorites. The liquid-vapor separator would operate at essentially the same temperature as that at the turbine inlet. The pump would be at the same temperature as the radiator. As indicated above, the alternator will probably be operated at its maximum operable temperature, hopefully, 900° F or higher.

The fluid pressure in the power system will depend on the working fluid and temperatures chosen for the system. The turbine inlet pressure would probably be between 80 and 250 psia and the condensor inlet pressure would be between 5 and 30 psia.

(b) Ion Motor

For cesium as the propellant, the supply tank would have to be maintained at about 100° F and the vaporizing chamber at about 1,400° F. The temperature of the ionizer would depend on the material used but would probably be in the range from 2,000° F to 2,600° F to maintain the desired ion current. The accelerating electrodes would be heated by radiation from the ionizer to a temperature approaching that of the ionizer. Thus, the electrodes may have to be cooled since the choice of electrode material may have to be made on other grounds than high temperature capabilities, e.g., resistance to high voltage breakdown or sputtering by ions.

The propellant flow rate would be quite small, of the order of 10^{-4} lb/sec.

(3) Environments

Various components of the power conversion system will be subjected to nuclear radiation from the reactor. The flux density will depend on the vehicle configuration and on the amount of shielding used. The flux density at the ion motor would probably be small due to attenuation by intervening components.

The ion rocket would be a space vehicle and thus would be operating in a vacuum and be subjected to bombardment by meteorites; molecules, atoms, and atomic particles; and solar radiation. The number, size, and velocities of the particles or the intensity and frequencies of the radiation which would strike an ion rocket are difficult to estimate on the basis of present data. Also, they are apparently not constant in space and will vary with such variables as proximity to the sun and the advent of solar flares.

(4) Potential TPS

(a) Discussion of Various Solutions

Since the ion rocket will be used in space, heat can be rejected only by thermal radiation. Because of the long flight times involved, thermal equilibrium will be attained. Certain components such as bearings, generator coils, and propellant supply tanks may have to be cooled but, again, the heat will ultimately have to be rejected by thermal radiation.

The maximum temperature in the power system will be determined by thermodynamic cycle requirements and will be controlled by the reactor. The maximum temperature in the ion motor will be the temperature of the ionizer which must be maintained, by power from the power conversion system, at a high temperature for efficient ionization of the propellant.

(b) Recommended TPS's

Thermal radiation is the only feasible thermal protection method for an ion rocket operating in space.

(c) Required Materials and Properties

Probably the main criteria for the choice of materials for the radiator will be long term compatibility with the liquid alkali metal and ease of fabrication. The pressure in the radiator tubes will be quite small and, thus, the stress in the tube walls will be low and should not pose any great problems[redacted] the other hand, the radiator tube material will have to withstand meteoritic impact without puncturing unless this problem is circumvented through the use of a bumper against which the meteorites would expend all or most of their energy.

The emissivity of the radiator is of extreme importance to the minimization of radiator area. Most metals have rather low emissivities. The presence of an oxide coating will improve the emissivity, but it is somewhat questionable how long a natural oxide would remain on a metal in the extremely low pressures in space, at the temperatures at which the radiator would be operating. Thus a high emissivity coating should be developed for the radiator. Again, the coating should be capable of remaining in place without deterioration for long periods at radiator temperatures, in the high vacuums of space, while being subjected to bombardment by meteorites, solar radiation, etc.

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Heating of the radiator by solar radiation has not been discussed. At least at the higher radiator temperatures discussed earlier, the amount of solar radiation heating would be small compared to the amount of heat being radiated out. If the ion rocket would be operating close to the sun or if the cooling of certain components demanded a lower radiator temperature, then solar radiation heating could be a serious problem. In this case, selective spectral coatings must be developed. These are coatings which would have low absorptivity for radiation with wave lengths corresponding to those carrying most of the energy radiated by the sun and a high emissivity at other wave lengths.

(d) Testing Requirements

The emissivities of potential radiator metals and coating materials should be determined at the temperatures at which the radiator would be operating. Investigations should be conducted to simulate the effect of the combined thermal and space environments on metal surfaces and coatings. At the extremely low pressures of space and at the temperatures in question materials will probably sublime. Even if the rate is not high enough to destroy the surface, the emissivity could be altered. Then too, bombardment by meteorites, atoms, molecules, X-rays, etc. would destroy the surface or, at least, affect the emissivity. In this connection, more data is needed on the environment to be encountered in space.

(5) Recommendations

The only thermal protection system which can be used in an ion rocket which is to operate in space for long periods is a thermal radiation system. Investigations must be conducted on (1) the long time compatibility of possible radiator materials with liquid alkali metals,

(2) the resistance of radiator tubing or bumper materials to penetration by meteorites, and (3) high emissivity and, possibly, selective spectral coatings which will not deteriorate at radiator temperatures in a space environment.

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PART III - TESTING REQUIREMENTS

1. MATERIALS TESTING AND ENVIRONMENTAL SIMULATION

The essential parameters which provide the basis for a materials
test depend to a great extent on the application. A good materials
test, however, need not provide the detailed scientific basis of the
phenomena being measured nor is it reasonable to expect complete environ-
mental simulation. A test that economically provides one design criterion
or exact knowledge of one property is generally more desirable than one X
which measures the integrated performance of materials design, fabrica- 6
tion and operational variations. There is no substitute for scientific 5
knowledge and no substitute for operational tests. It is just as unrea- 0
sonable to make all of science into engineering or to provide detailed
engineering criteria from simulation or operational testing. Very little
controversy can be generated with regard to the validity and usefulness
of the tensile test and its value as an engineering test. One does not
confuse tensile testing with the intricacies of solid state physics or
load redundancy in structural or flight test.

Investigations of ablation phenomena have led to extensive discussion and strong opinion in two areas (1) chemical and physical mechanisms of ablation, and (2) the accuracy of flight simulation and the pertinent measuring techniques. Both areas have made valuable contributions to the requirements of a sound materials test. On the other hand, the details of chemical kinetics and similarity parameters have tended to confuse the specification of an acceptable materials test.

The objective of this discussion is to establish the basis for useful thermal protection materials tests, however controversial they may be. No attempt will be made to satisfy all disciplines equally since that perhaps is not possible nor even desirable. Emphasis will be placed upon producing material data useful for future aerospace vehicle design where thermal protection is required. Furthermore, all types of thermal protection will be considered, not just ablation or transpiration processes. Every attempt will be made to take maximum advantage of existing progress in theoretical analysis and experimental techniques without unnecessarily involving a materials testing with theoretical uncertainty.

Present and Required State of Art

It is an accepted fact that a substantial quantity of uncertain materials information has been generated with respect to thermal protection, particularly in the area of ablation materials. Much of the confusion is the result of inadequately informed materials personnel applying materials to heat sources of unspecified characteristics.

Little consideration was given to what materials data were essential or existing knowledge in other areas, such as aerothermodynamics. At the same time available information was scarce and in justice to these early results, real progress was realized in compiled information and even through mistakes.

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There is some doubt, even now, as to whether an acceptable "standardized" materials test can be defined. This is due to uncertainties in the measurement of certain test parameters or inadequate experience with tests described in the literature. Additional investigation will be necessary in order to establish completely acceptable, accurate measurements both on the material and the test conditions. However it is now possible to specify those parameters which define a good test and to provide a plan which will lead to comparable materials data. It is further possible to specify the materials properties and characteristics which define the thermal protection capability and to at least specify (if not completely define) the test measurements.

It is not intended here to outline or design certain tests. Such standard tests can only be established by means of a well planned development program, with participation of the majority of laboratories engaged in thermal protection testing. Instead, it is attempted in the following to discuss the significance of the essential testing parameters in some detail. This is augmented in Appendix A by the description of potential standard test methods as they have been developed by a leading laboratory and so far used successfully. No judgement is passed on the proposed methods and it is left to the reader to appraise their validity and applicability in the light of the following general discussion.

2. TEST ENVIRONMENTS

a. Environmental Characteristics of Test Devices

In pace with the development of TPS, a variety of testing devices has been designed and used with varied success. They all attempt to simulate the flight environment, which is basically composed of four major elements: the application of heat (enthalpy), the mass flow (including its mechanical effects), the gas composition (air) and time (of operation).

Solar and Arc Image Furnaces provide a satisfactory heat flux, yet are limited with respect to temperature. The mode of heat transfer is purely radiative. Influences of gas chemistry can be accurately investigated, while mass flow effects cannot be reproduced.

Rocket Exhaust Jets furnish a satisfactory representation of heating and mass flow conditions and can be operated for prolonged times.

They fail however in the reproduction of the gas environment (combustion gases instead of air).

Combustion Torches have characteristics similar to the jet rocket exhaust, yet are obviously very limited in heat input and mass flow. They are confined to very superficial screening tests.

Pebblebed Heaters produce a fairly accurate representation of mass flow and gas chemistry, yet are limited with regard to enthalpy and operation time. They are excellent tools for tests in the moderate heat flux-limited time field.

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The Shock Tubes produce undoubtedly the most accurate simulation of the aerothermal environment. They are however not useful for materials tests due to their obvious time limitations.

The Constricted Arc or Plasmajet can produce high enthalpies and moderate mass flow rates for prolonged times; by the use of air or the combined use of nitrogen and oxygen a satisfactory simulation of the gas chemistry can be obtained. Plasmajets lend themselves also to the simultaneous simulation of convective and radiative heat transfer by the use of radiant shrouds or the combination with an arc image device.

It is apparent that all these devices have certain shortcomings in the true representation of the flight environment. In the search for generally applicable materials tests, preference should be given to the device which produces an optimum compromise between true flight simulation and adaptability to materials testing, and at the same time is convenient and not too expensive. It appears to be generally agreed between the experts in this field, that the constricted arc or plasmajet provides the best compromise.

While there are numerous types of plasmajet facilities which have individual desirable and undesirable characteristics, adequate power, mass flow and intelligent design can produce a reasonably acceptable test tool. Some of these are discussed in Appendix A. The individual characteristics of various forms of plasmajet facilities will not be discussed in this commentary. Neither will any attempt be made to judge the accuracy of aerothermal simulation. The purpose of this discussion is to suggest what may be necessary as a materials test device, the energy conditions for an acceptable materials performance test and the types of measurement which should accompany it. While most of the comments apply to ablation and transpiration requirements, the same facility will be extremely useful for the comparison of other TPS and other such problems as corrosion in high temperature high velocity air streams. Primary emphasis will be placed on those variables which affect the materials being tested rather than the mechanics of operation.

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b. Discussion of Individual Test Parameters

(1) Heat

Heat is the primary element of a TPS materials test facility. This energy must be available in the form of a moving gas stream. It should be sufficient to specify how much heat should be available to cover the range of required materials test and the form in which it should be available.

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(2) Enthalpy (H)

This thermodynamic value is defined by the following equation:

$$H = q + Apv$$

where

q internal energy (atomic or molecular)

p pressure

v volume (specific)

A mechanical heat equivalent

While the pv term is critical in the re-entry, it is not in the test case. In the test, the enthalpy is almost totally determined by the heat content of the system, since physical limitations dictate this condition. This is not unusual and is widely used as the constant pressure conditions of chemical thermodynamics. Therefore the enthalpy difference frequently used to correct the cold wall enthalpy difference is nothing more than the heat capacity differential of the specimen and the boundary layer.

Materials test facilities should have an enthalpy capability between 1,000 and 15,000 Btu/lb. Futhermore no continuous variation is required, but variation within steps of the total range is most convenient. In all cases a uniform test area must be available.

(3) Enthalpy Measurement

There is a substantial degree of disagreement regarding the measurement of enthalpy and heat transfer rate in plasmajets. There is little to be gained in discussing the controversies. Several techniques such as melting point calorimeters, cold calorimeters, combinations of calorimeters

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and pressure or thrust indications, (total enthalpy probes) and plasma-jet heat balance measurements are available. Additional work is necessary to establish complete agreement but probably not as necessary as the abandonment of pet techniques and theories. The cold wall calorimeter which occupies (including guard) less than one-half the radius of the jet is a reasonably good approximation for an acceptable measurement of heat transfer rate under present conditions. It also appears reasonable to define thermal capacity (Q*) in terms of cold wall heat flux measurements. Two points are recognized: (1) there are other important variables which influence Q*, as pointed out in Appendix A, and (2) heat flux is not enthalpy and the thermal capacity is a function of the enthalpy level. This relationship as a test facility may be expressed by

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$$\dot{q} = H \cdot \dot{m}$$

where

$\dot{q}$ heat flux

H gas heat content

$\dot{m}$ mass flow rate

For such a relation it is obvious that the same heat flux can be obtained from a number of H and m values. A very high heat flux can, e.g., be measured in a low temperature gas with a very high mass flow rate.

(4) Uniformity of Heat Application

A dynamic gas facility must provide a reasonable uniform cross sectional area with respect to heat content and flow. How this is obtained is not important, but at least 2/3 of the axial portion of the test medium should have less than 5% variation in total enthalpy. Uniformity with respect to composition and contamination will be discussed in a later section. It should be mentioned that few commercially available or home grown plasmajets have such uniformity without stilling chambers and secondary expansion nozzles.

(5) Radiant and Non-Equilibrium Heat Transfer

There are many problems with respect to contributions from radiation components in the boundary layer, the electrical discharge and the gas. Under present circumstances these problems tend to become exaggerated, except for hyperbolic re-entry conditions and shroud nozzles. However if some of the present variables can be controlled there will be a much greater possibility of solving future problems.

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At this writing radiation heat transfer and non-equilibrium discrepancies do not affect heat transfer measurement by more than ten percent in intelligently engineered facilities. Furthermore, since this tends to be a constant in a particular set of comparative tests, it becomes even less controversial.

(6) Mass Flow

This is the most clear cut and easiest to measure variable in all plasmajet operation. A meterials test requires that mass flow with the X650 resultant pressure and shear forces be rather high. This of course depends upon many conditions and engineering characteristics of the test apparatus. Buhler and associates at Plasmadyne Corporation have made notable progress in calibrating gas dynamic parameters. Their work should be carefully reviewed for standard techniques of pressure measurement and relation to simulation accuracy.

The control of mass flow by stilling chambers, auxiliary nozzles including shroud nozzles and choking is entirely dependent on the immediate requirements of a test. Since it is good practice to measure total and dynamic pressures, little can be said with regard to a materials test beyond this point. Mass flow rate requirements are therefore variable, but should fall between 0.005 and 0.1 lb/ft^2sec. Flow must be supersonic for an acceptable materials test.

(7) Gas Composition and Contamination

The composition of the test gas in measuring ablation rates and similar phenomena must be 78% N_2 and 22% O_2. This is usually accomplished by operating the plasmajet with nitrogen and post-mixing sufficient oxygen to make the above specified composition. Low pressure ($<$0.01 atmospheric) plasmajet may be run on air without extensive electrode contamination, provided water cooled metallic electrodes are used.

The subject of contamination in an aeroplasma can be approached with justifiable misgivings. In general contamination has no place in an acceptable test environment, excuses suggested by reason of improper design, operation or materials selection notwithstanding. The use of graphite parts in a materials test plasmajet is intolerable. Cooled metallic electrodes are not foolproof because poor design or improper operation can contribute equally objectionable contamination. Generally speaking contamination should be less than 0.1% per gram atom weight.

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3. SAMPLING CONSIDERATIONS

a. Sample Size

The factors that frequently control sample size are the test conditions available, cogently described in Appendix A as being "according to budget, available power supply and desired environment - in that order." Adequate facilities are available for accurate measurement of the materials behavior as applied to radiation, insulation and convective cooling systems. Therefore most of the comments made here will concern the testing of absorptive TPS in dynamic hot gas facilities. X
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The first criterion of materials sample size both in surface area 5
and thickness is that it be compatible with the structure of the material. 0
This applies particularly to reenforced plastics, laminates and composite structures where geometry, fiber orientation, and directional properties of the sample must be representative of the engineering application.

It is debatable as to whether or not the jet dimensions should influence sample size. There are more reasons why the sample should be totally immersed in the gas stream than there are for partial sample exposure, even for flat 180^{o} geometry "splash exposures." Thermal diffusion effects, non-uniform jet characteristics, weight loss errors and many other factors argue against partial sample exposures. With recently developed pipe and shroud techniques it seems reasonable that a one inch diameter sample could be exposed to satisfactory gas enthalpies at moderate power investments and with generally safe sampling statistics.

b. Sample Geometry and Mounting

Most of the general comments made with respect to sample size also apply to sample geometry. However mounting presents a special problem that should be discussed under this heading.

Flat face cylindrical samples with mechanical feed represent the most convenient and probably the most useful geometry for materials testing. It provides an excellent geometry for thermocouple mounting, rate measurements and surface study. If the sample is of the order of 1/3 of the jet cross-sectional area, fairly uniform stagnation point conditions are obtained. In well calibrated facilities this area ratio might be increased. Flat samples do have problems associated with down stream gas flow and heat transfer to the sides of a sample, whether it be cylindrical or rectangular. Guarding against side heat flow can be a meticulously careful experimental technique with few admirers and a multitude of critics. Finally the flat face geometry is not ideally suited for char stability studies.

Other geometries such as conical, spherical or contoured, must be considered for more complete materials evaluation. In all such cases materials engineers would be well advised to seek the assistance of thermodynamicists, aerodynamics or the more recently conceived hybrid of these species, the aerothermodynamicist. Flow conditions and heat transfer rates with relation to the properties of unperturbed flow can cause serious errors in an otherwise straightforward experiment. Since such geometries are usually scaled down from full scale configurations, the necessary establishment of scaling laws for the heat transfer can become very complex. It has been generally observed that heat transfer and flow conditions down stream from the stagnation area can be at least as severe as that measured at the stagnation point. When materials are tested in specific configurations, a calorimeter of the same size and shape may be indicated to measure total cold wall heat input. These are among the many difficult areas of question that require increased experimental verification. Specific shapes are, however, essential to study effects of shear, char tenacity, radiation distribution, thermal shock, pyrolitic gas effects and the behavior of molten ablation products. The use of nozzles or shrouds, which have been contoured for specific sample geometry and required flow properties, may be advantageous. The distribution and control of radiant energy over the sample as affected by the radiation of uncooled extension nozzles or shrouds, is presently under study.

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The mounting arrangements particularly where radiative-insulation, heat sink, convection cooled and long term ablation tests are conducted in gas dynamic facilities suggest a few special words of precaution. The importance of lateral heat transfer both in flat plate and other geometries has been pointed out. An equally serious error in the measurements may result from thermal conduction and capacity in the mounting and attachments during tests requiring more than one minute of exposure. This may be aggravated by careless selection of mass ratio of sample to holding fixture. Low conductivity, low heat capacity, low mass fixtures or the most desirable, calibrated heat leak fixtures can overcome this source of error.

c. <u>Sample Positioning</u>

Sample position during test is dependent on a number of factors most of which are difficult or impossible to define. At least the sample should be positioned in an area which is experimentally calibrated and not subject to casual variations. The degree of radiant energy exposure on the sample from the cathodes of direct current plasmajets is a controversial subject difficult to resolve or correct. It could be the source of large errors particularly when organic ablative samples are tested and a convective-conductive heat transfer environment is expected. While not generally available, right angle flow may be produced in plasmajets, a technique which needs additional experimental development.

4. SIGNIFICANT MATERIALS PROPERTIES AND DATA

a. Design Considerations

The use of any material in engineering or design application demands that certain properties be established and that the statistical variations of such properties be known. These data are designated as design allowables. For any specific application a number of mechanical and physical properties must be known and assessed with respect to operational requirements and the anticipated environment. This deductive process is then followed with component fabrication and preliminary simulation testing. From the results of simulation tests and the materials design data some correlation can be derived although it is never exact. Even less correlation between full scale simulation and materials properties can be expected.

The primary requirement for thermal protection design is accurate materials information. There is every indication that this could be available once it is understood which material properties are representative of service behavior and what specifications are necessary in a test to reproducibly and economically demonstrate these properties.

This section discusses various properties applicable to thermal protection performance. Agreement must be reached in expressing and standardizing these allowables or very similar ones if any rational approach is to be anticipated. Considerable efforts and funds have been spent and will continue to be spent in poorly defined testing procedures which produce mountains of conflicting and unresolvable information. There is a serious obligation on the part of all concerned, and in particular on the part of the government and industrial contracting agencies to implement a program which will lead to the essential uniformity of test and the publication of data.

b. Primary Materials Properties

The application of a material or materials composite in a thermal protection system has one basic requirement - the ability to absorb or impede the flow of heat energy. Since matter has a limited energy absorption capacity, the weight or mass and the rate of energy absorption or impedance must be known if any practical or engineering use is to be anticipated. From this may be derived the design criteria which govern the use of thermal protection materials;

(1) Thermal capacity

(2) Weight

(3) Time or rate characteristics

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When matter becomes involved with an energy transfer or storage phenomenon, the capacitive factor must be accompanied simultaneously by an intensity factor. In this case the driving potential is the enthalpy difference between the energy source (the boundary layer) and the receptor (the material). This intensive factor for the problems considered is more appropriately discussed in connection with test criteria and measurement aspects. It is mentioned at this point for the purposes of technical accuracy and to emphasize that the thermal capacity is always dependent on the intensity factor. Engineering practice both in thermal, electrical and stress applications (e.g., tensile strength) generally considers average capacity over a limited or specified range of intensity. In accurate analysis the relationship of capacity versus intensity has to be established, with other factors held as parameters. Recognizing this, the practical requirements of the materials engineer then dictate the necessity of establishing a fixed point (thermal capacity) so that other constantly changing variables can be assessed.

If the so accepted thermal capacity concept can be recognized as a materials and engineering property, then associated conditions can be specified which make it more meaningful. If a standard test is to be obtained, and certainly there is agreement regarding its desirability, some representative value must be carefully defined as well as a test procedure which is practical and reasonable. It is recommended therefore that the "thermal capacity" be accepted as the representative value as it comes closest to the needs of the design engineer. This is more desirable than "heat of ablation," because the latter term is too limited and does not permit direct comparisons with other forms of thermal protection systems. Furthermore many of the proposed definitions, while technically accurate, relate the engineering quantity to the test conditions rather than the application. A very minor modification, the introduction of weight per unit of surface area and the use of time as a parameter in the numerator produces just as accurate a presentation and makes the thermal capacity term relatable to design parameters.

Insulation and radiative systems are special cases of thermal protection systems for several reasons:

(1) Radiative systems do not involve weight since they are based on a surface phenomenon. However if the radiative capability or lack of it in the sample will be indicated by the required magnitude of thermal capacity or the required weight of thermal protection, this does not relieve the requirement for establishing the sample emissivity or surface temperature during test.

(2) Insulation systems, even when the outer surface is in radiative equilibrium with the environment, have a finite thermal conductivity or diffusivity. The magnitude of this thermal leak and the permissible back face temperature establish time of operation and the weight of

insulation required. When these factors are known, direct weight comparison may be made with absorptive type thermal protection systems. Indeed the low rate long term ablation system must be examined in a similar manner because, over extended periods of time, the thermal diffusivity of the ablation material must be known. Finally, it has been demonstrated that radiative insulation systems can be screened or assessed by the same techniques generally used for absorptive systems. However, established thermal conductivity techniques are usually more convenient and more accurate.

c. Secondary Properties and Characteristics

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Aside from those primary variables in a materials test, there are many other factors of a secondary nature which must be recognized. The more important of these are

- Thermophysical properties
  - (emissivity)
  - (specific heat)
  - (thermal conductivity)
  - (heats of reaction and change of state)
- Molten viscosity
- Char formation
- Shear and compressive strength
- Thermal expansion
- Sonic fatigue
- High temperature corrosion
- Behavior in vacuum and/or ionizing radiation
- Density and weight
- Pyrolytic gas properties (physical and chemical)
- Fabrication variables

These properties or characteristics do not affect all materials equally, nor are they of equal importance. Some have a critical bearing on test procedures, while others will vary as a result of the test procedure. Each will be discussed in proportion to its influence on the reliability of an acceptable materials test.

(1) Thermophysical Properties

This section refers to those vital physical properties which control the behavior of materials at high temperature. While some may be questioned as thermophysical properties they are included in this discussion because of related phenomena or measurement techniques. Those properties or behaviors which will be considered are, emissivity, thermal conductivity, specific heat, heats of reaction vaporization and/or sublimation spectral absorptivity, thermal expansion and pyrolitic gas properties.

While is is impossible to rate these phenomena with regard to relative importance, emissivity doubtless plays a significant role in any type of thermal protection. It is a critical property not only for radiant conditions but is of equal importance in rejecting heat that is accumulated from convective and conductive energy transfer. The measurement of surface emissivity is a most delicate procedure even under very closely controlled laboratory conditions. The measurement of surface emissivity during ablation testing is therefore not a recommendable technique. A measurement of a char layer after test may be possible but difficult, particularly at elevated temperature. For practical
X purposes, comparative surface temperature or radiometric measurements
6 under calibrated environmental conditions with carefully calibrated
5 emissivity standards appear to be the most obtainable technique pres-
0 ently available. Associated with the correct measurement of surface temperature and emissivities are the associated problems of spectral absorption both on the surface in the matrix material and transpirated gas properties which cover the surface of the sample. While these are not primary measurements they influence other measurements, the mechanism of energy transfer and materials behavior. They should not be confused with primary measurements but must be placed in the category of necessary research.

The more commonly known thermophysical properties such as thermal conductivity, specific heat and thermal expansion are vitally essential. They must be separately measured and made available with the primary data of thermal capacity and performance. As previously mentioned, these data are significant for thermal protection systems involving insulation, convectively cooled and heat sink characteristics. They become almost equally important when ablation or transpiration are considered for longer duration. There is an increasing tendency to measure thermal diffusivity α where

$$\alpha = \frac{k}{\rho \cdot cp}$$

where

k thermal conductivity

ρ density

cp specific heat

It is desirable and, under some conditions, may be possible to measure this property during a dynamic test.

Heats of reaction, vaporization and sublimation are extremely useful for kinetical interpretations of the ablation and transpiration.

Such measurements must be made in research facilities, as previously distinguished from facilities.

(2) Molten Viscosity

There is no established concept for the significance of this phenomenon which is observed on the surface of ablation specimens. It may find a possible application in the assessment of shear forces acting on a specimen but should not be awarded a high priority except in very basic studies of ablation phenomena.

(3) Char Formation

Surface carbonaceous material which forms during the ablation of organic base composites has an important bearing on the overall heat absorption characteristics. Its stability depends upon the materials properties and the mass flow conditions of the environment, in service as well as in a test. Removal of char by poorly designed test conditions leads to erroneous results just as does the erroneous assumption of char stability derived from tests at low mass flow conditions. Since it is not possible to exactly simulate operational conditions in a laboratory materials test, this characteristic should be independently assessed either by separate laboratory examination, or by exposure in a high mass flow rate facility (e.g. rocket exhaust) where environmental pressures and shear forces can be reproduced.

(4) Shear and Compressive Strength

These are mechanical properties which should be independently measured and used as guides to test evaluation and to design requirements. Shear stress should be evaluated over a range of surface temperatures, since like in char layers, surface cavitation and spalling leads to large errors in thermal capacity. These properties can be of major significance in lightweight high efficiency insulation systems.

(5) Thermal Expansion

For certain thermal protection systems thermal shock behavior must be carefully assessed. For this reason thermal expansion properties combined with tensile properties should be established as supplementary information.

(6) Sonic Fatigue

The behavior of materials under severe sonic vibration is a somewhat new consideration for thermal protection systems. While it applies predominantly to long time, stable systems with metallic skin (e.g. high efficiency insulation panel) it may also apply in the future to other systems such as convectively cooled, long duration ablation and transpiration systems. Boundary layer noise up to 150 db may be expected at velocities in excess of M 5 at altitudes exceeding 100,000 feet. The detailed effects of this phenomenon on hot gas environments, or its presence in supersonic hot gas facilities are not known. It is possible that such an environment could change heat transfer characteristics in the boundary layer.

(7) High Temperature Oxidation

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Very little quantitative or even comparable data are available with regard to oxidation rates of metals or organics under moving air and thermal cycling conditions. The problem is most acute for insulation panels which must operate in excess of 1500° F for long periods of time with cyclic temperature changes. While considerable activity is now in progress in this area, there exists an equal need for standardization and uniformity of tests and the expression of results.

(8) Vacuum and Ionizing Radiation

The subject of space environment and its effects on thermal protection systems of all types would demand an extensive study. There are many possible effects and many combinations of environments which might be discussed, which are beyond the scope of this report. While these areas could have an important influence on thermal protection behavior, they are secondary to the immediate goal of an acceptable materials test. Once this is accomplished, the influences of other environments can be investigated with more intelligence. It is sufficient to point out the potential problem and to suggest careful consideration in those operational areas, where long periods of time must be spent in space prior to the use of a thermal protection system.

(9) Density

This fundamental physical property as well as the total weight change should require little elaboration. It is the most obvious materials property to be measured and, while basically simple, can be subject to procedural errors. Some early ablation weight losses were based upon loss in weight after the char layer had been scraped from the sample; others were not. In almost all cases it is exceptionally difficult to avoid sample losses after ablation testing and provide a technically accurate weight loss figure. In fragile char layers and refractory oxide, bubbles are formed during ablation, and it is difficult to define a satisfactory measurement system. A development project, which would provide a means for indicating weight changes during tests, would be extremely useful.

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(10) Fabrication Aspects

The appraisal of this subject is well beyond the scope of this report. It is recognized and little change can be expected in the near future. Closer specifications in the areas discussed in this section are desirable. Plastic testing procedures are now under agressive standardization programs. For the near future however these variations can only be normalized and treated statistically. In intelligently run materials test programs they should constitute a minor error compared with those now being tolerated in test procedures.

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PART IV - CONCLUSIONS

1. STATE-OF-THE-ART SUMMARY

The ultimate objective of this report is to define the most significant TPS, and to recommend a degree of effort which should be placed on the related materials R&D to meet the future requirements of NASA. To arrive at an objective judgement, three considerations have to be weighed against each other: the requirements for thermal protection in future NASA projects, the capabilities and growth potential of TPS's which satisfy these requirements, and the magnitude of the present R&D effort. The degree of future R&D effort should be proportional to the requirements and the potential of the TPS, and inversely proportional to present efforts: Systems which have little potential should not be pursued, regardless of requirements and present efforts. Likewise, efforts should be held at a minimum level, if no requirements exist or are foreseen. Primary emphasis should, however, be placed in these areas, where both requirements and potential are high. The degree of effort depends then on the magnitude of present effort. By the same token, a reduction of the present effort is indicated, if it is higher than either requirements or potential justify.

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The overall development status of TPS's is summarized in Table 6.

The potential of direct radiation systems is too limited to produce any substantial gain over their present capabilities, except for solar heating in space, where they constitute the primary tool of temperature control. Even though their potential is likewise limited, the simplicity of the heat dissipation at the place of heat input warrants high efforts to be placed on the development of coatings with balanced emissivities. There is a vital need for increased emphasis in the following areas as applied to surfaces for space craft temperature control:

(a) Development of surfaces with reproducible emissivity characteristics - particularly the specular metallic surface.

(b) Development of standardized measurement techniques and standard surfaces.

(c) Development of surfaces capable of withstanding the various prelaunch, as well as space environments, with little or no change in emissivity characteristics.

Another proposed system for protection against extreme heating rates and prolonged time periods is to repel the ionized gases with magnetic fields (magneto-hydrodynamics).

Conventional insulation has arrived at a saturated state of development, where further efforts will only produce marginal improvements. It has become a part of straightforward engineering and requires no further research efforts. However, the extended durations of flight at intermediate heat flux levels, encountered in advanced flight and manned re-entry vehicles, create severe requirements for an adequate passive TPS, which operates without functional means. Advanced, high efficiency insulation has a promising growth potential, and highest efforts, particularly fundamental research on the means of controlling the heat transfer characteristics of materials, are urgently needed.

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TABLE 6.- DEVELOPMENT STATUS OF TPS

TPS	Potential	Design Requirements	Present Development Efforts	Recommended Priority
Direct radiation:				
Aerodynamic heating	-	-	-	-
Space (solar)	+	++	+	+
Insulation:				
Conventional	o	o	+	o
High efficiency	++	++	o	++
Indirect dissipation:				
Aerodynamic heating	+	+	o	+
Space (solar)	++	++	o	++
Convective cooling:				
Circulating	o	+	o	o
Heat sink	-	o	+	-
Ablation:				
High $\dot{q}$	+	+	++	o
Moderate $\dot{q}$	++	++	+	++
Film cooling:				
Direct	-	+	-	-
BL gas injection	++	++	o	++
Transpiration	++	+	-	+
Energy conversion	++	++	-	++

Legend: o average
- below average
+ high
++ extremely high

The potential of indirect dissipation has so far been very little evaluated. There is, on the other hand, a definite need for temperature reduction at certain "hot spots" in advanced flight systems (e.g., leading edges) without functional means. Increased R&D effort is indicated. An extremely high application potential exists in advanced space vehicles, where direct radiation is insufficient for temperature control. Highest R&D efforts are in order.

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The application of convective cooling is limited in view of the extreme weight penalties, to requirements which call for an unlimited time of operation at heat flux levels which cannot be met by passive means. No particular R&D efforts appear to be justified.

In view of the emphasis placed on heat sink systems in the initial development of ICBM's its capabilities have been extensively explored and appraised. Continued development of heat sink systems is not likely to produce useful results for NASA missions, whatever the benefit may be for ballistic missiles, except for the potential applications of pyrolytic graphite.

Likewise, a high state-of-the-art has been achieved in high heating rate ablation systems, as they apply particularly to the steep (ballistic) re-entry. The potential of ablation at moderate heat flux rates and long operation times, however, is insufficiently explored, even though extreme requirements are apparent. A reduction of high heating rate efforts in favor of R&D in the intermediate heat flux-time field is in order. In addition, there is a definite need for increased efforts in the application of ablation to solid propellant rocket nozzles.

The very limited capabilities of liquid film cooling, paired with considerable weight requirements, do not warrant any particular efforts. Gaseous film cooling by gas injection in the boundary layer, however, appears to have considerable growth potential, particularly for applications in the most problematic intermediate heat flux-time field. With the exception of NASA work (Ames), this promising system is greatly neglected. A high R&D priority should be assigned.

Transpiration systems have considerable potential for prolonged periods of operation at extreme heating rates. Even though no requirement exists or is foreseen for this condition, a minimum state of the art should be maintained. As this field is completely neglected at present, adequate efforts should be initiated.

The application of direct energy conversion materials and techniques to the absorption of heat is in a very early feasibility study state, yet is extremely attractive for space applications, where utilization of the otherwise undesirable heat would be essential. A high rate of R&D effort is indicated.

[redacted]

One of the major difficulties in the development of TPS is the lack of pertinent materials data. In the first place, even conventional properties of materials are not available in certain high, as well as low temperature ranges. Second, the behavior of matter at extreme temperatures and the transition to the thermal plasma are not sufficiently explored, nor the interrelation of phenomena understood. Research in these two areas is indicated, particularly in view of the time requirements of such fundamental investigations.

Of immediate concern is, however, the establishment of performance
data on TPS as well as related materials. Most of the few numerical data X
available are not comparable, as they were obtained under a variety of 6
more or less defined environments. The general acceptance of a standard 5
test method, standardized samples and environments are a prerequisite of 0
any future development work on TPS. Immediate efforts should be placed in the development of a standard test method and a uniform representation of test results.

2. REQUIREMENTS OF NASA VEHICLE PROGRAMS*

The present as well as expected future vehicle program of NASA indicates the following four primary requirements for thermal protection:

(1) Drag and lift surfaces of return vehicles from orbital and interplanetary operations, related to the intermediate flight regime. Two heating characteristics are encountered.

(a) Prolonged (4-60 min) atmospheric exposure to intermediate heat flux levels (15-250 Btu/ft^2 sec), as encountered in high velocity cruise or during return from orbital operations.

(b) Two-cycle atmospheric heating, consisting of a short (1-6 min) exposure to high heat flux levels (250-800 Btu/ft^2 sec), followed by an exposure similar to (1), as encountered in return maneuvers from interplanetary operations.

(II-2)

(2) Solid propellant rocket nozzles and casings, subjected to heating by various types of combustion gases and products, with heat flux rates up to 500 Btu/ft^2 sec and times from seconds to several minutes.

(II-4)

(3) Temperature control in the hulls and heat exchange elements of space vehicles, exposed to radiant solar heating, in near space ($\dot{q}$ = 0.12 Btu/ft^2 sec) and deep space (preferred range 0.24 to 0.052 Btu/ft^2 sec).

(II-3)

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*The numbers in parentheses at the end of paragraphs indicate the related report sections. [redacted]

[redacted]

(4) Protection of space vehicles against the heating from power-plants and propulsion systems, exhibiting a variety of temperatures up to 1,500° F and heat flux levels up to 36 Btu/ft^2 sec.

(II-4)

3. RECOMMENDATIONS*

Based on NASA vehicle requirements and the potential of various TPS, primary materials research effort is recommended in the following fields:

(1) Development of lightweight "high efficiency insulation" systems, which substantially increase the thermal capabilities of passive thermal protection for unlimited operation time, as required in high speed cruise flight and orbital lift re-entry. Development should be focused at the heat flux range from 10 to 50 Btu/ft^2 sec.

(I-3c(2);I-5)

(2) Development of low rate ("slow") ablation systems for the intermediate flight regime, with heat flux levels from 15-250 Btu/ft^2 sec and times from 4-60 minutes. The development should include the study of the downstream effects of ablation products.

(I-3d(2);I-5;II-2)

(3) Two-cycle ablation systems for return maneuvers from space operations, compatible with space environment before as well as between cycles, and capable of heat absorption as follows:

Cycle I: $\dot{q}$ = 250-800 Btu/ft^2 sec   Cycle II: $\dot{q}$ = 15-250 Btu/ft^2 sec

t = 1-6 min   t = 4-60 min

(II-2)

(4) (a) Studies and experimental investigations on the radiative heating from the hot boundary layer, and particularly from the hot gas cap between the vehicle nose and the detached bow wave at parabolic velocity and (b) development of TPS for combined convective and radiative heating, as established in (a).

(II-2a(1))

(5) Development of boundary layer gas injection systems for gaseous film cooling, applicable primarily to the intermediate heat flux field (15-250 Btu/ft^2 sec) and prolonged time of operation, as encountered in high speed atmospheric cruise and space return vehicles. Besides the primary objective of thermal protection, its aerodynamic effects (drag) should be investigated.

(I-3d(4);II-2)

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*The numbers in parentheses at the end of paragraphs indicate the related report sections.

[redacted]

(6) Development of coatings compatible with space environment, with defined ratios of absorptivity to emissivity, for temperature control of space vehicles. Two heating characteristics are encountered:

a. Constant heat flux of 0.12 Btu/ft^2 sec (near space)

b. Varying heat flux in interplanetary operations

(I-2d;I-3c(1);II-3)

(7) Development of indirect dissipation systems

a. For intermediate heat flux levels (up to 45 Btu/ft^2 sec) in limited areas of drag elements, such as leading edges or nose cones of cruise vehicles. X650

b. For temperature control in space. Conditions same as in (5) above.

(I-3c(3);II-3)

(8) Development of surface coatings capable of heat absorption and utilization by application of direct energy conversion materials and techniques, for the use in space vehicles. Systems have to be compatible with space environment.

(I-3d(5))

(9) Development of high efficiency insulation systems based exclusively on reflection and radiation for the use in power plants of space vehicles.

(II-6)

(10) A systematic investigation on the application of TPS to solid propellant rocket motors, to include:

Transpiration
Evaporation cooling
Chemical reaction cooling
Heat sink
Radiation
(Film cooling)
(Ablation)

(II-4)

(11) Maintenance of an adequate state of the art in transpiration cooling, with particular emphasis of long time operation at high heat flux levels (above 1,000 Btu/ft^2 sec).

(I-3d(3))

(12) Studies on the materials behavior at extreme temperatures, and establishment of thermal and other pertinent material properties at extremely high, as well as extremely low temperatures.

(13) Development and introduction of standard test methods for the establishment of performance data for TPS.

(I-4;III-2)

(14) Establishment of a list of standard terms, definitions, symbols and dimensions. Such a list is proposed in this report for acceptance.

(III-1)

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Commensurate with these proposed increased efforts, a reduction of efforts is recommended in the following fields, in which an adequate state-of-the-art has been achieved:

(15) Reduction of effects in heat sink systems, in view of their limited capability and the saturated state-of-art.

(I-3e(5))

(16) Reduction of efforts in short-time high heating rate ablative systems for ballistic nose cones, as all present and foreseen requirements can be adequately met with achieved capabilities. Remaining efforts should be directed toward a wider variety of potential material systems.

(I-3d(2);I-5)

(17) Reduction of efforts in conventional insulation, due to its obvious thermal limitations, which permit only marginal improvements.

(I-5)

(18) Reduction of present extensive efforts of theoretical treatment of TPS in favor of technological approaches, as outlined in (12) above.

(I-4)

NASA Headquarters,
Washington, D.C., October 1, 1961.

 [redacted]

APPENDIX A

TESTING TECHNIQUES, FACILITIES AND PROCEDURES

Most attempts to simulate high heat flux environments have employed arc plasma devices. There are as many testing techniques as there are units. The probability of standardizing is not very likely because the required details of environment differ from one test program to another and the units are often "home-made," or at least modified, according to the budget, available power supply and desired environment - in that order. One serious factor has been the exclusion of turbulent heating and the associated shear forces. In general, however, the failure to properly simulate the combined chemical-thermal-mechanical parameters leads to problems with the interpretation of results.

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With this in mind, the review summarizes a series of approaches taken in the past few years (primarily at one laboratory) in conjunction with re-entry vehicle materials testing. There were sufficient talents and funds available to allow for elaborate analytical studies and the latest of arc devices and techniques, with serious material problems that depended primarily on ground testing and only confirmation with instrumented flights. A great deal of production planning was based on the ground test results and extrapolations, and the results therefore had to be trustworthy or at least understood the limitations.

The prime purpose of this review is not to achieve standardization, but rather exemplify the experimental details and analysis which had to be taken into account. They illustrate the sophistication which had to be employed to avoid the much more time consuming, expensive gambling with full scale testing programs.

There is an abundance of tripe in the literature based on the philosophy of "have arc, will test," with "back-of-the-envelope" calculations and "profile comparisons." Testing of this type does not sufficiently resolve the TPS requirement to significantly improve the probability of designing and building hardware with reliable safety factors, meeting severe weight and size restrictions.

The review includes the following:

I. <u>Small Arc Techniques</u>

Three references (81, 82, 83) are given to published procedures and facilities covering early versions of splash test methods employing small arcs [40 to 100 kw in (81 and 82) and 300 kw in (83)].

[redacted]

II. Large Arc Techniques

A. The 500 kw arc facility for splash tests (Avco "500" Arc)

B. The 10 megawatt arc complex

1. Splash technique
2. Pipe technique
3. Shroud technique
4. Hyperthermal wind tunnel

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The small arc tests are generally inadequate, but since they are common laboratory items, they are the ones most likely to be misapplied and must be looked at most cautiously. This is obvious from the footnotes summarizing comments made by critical reviewers,[*] indicating questionable areas. Certainly any reference to standardization can only apply to an established "in-house" technique.

The larger facilities outlined in Part II are the most recent tests. The 500 kw splash tests and the 10 megawatt pipe tests are currently being employed on a routine basis. The shroud and the hyperthermal devices are still in the comparatively early stages of operation.

1. SMALL ARC TECHNIQUES

a. Description of Standardized High Temperature Material Systems Screening Test[1]

Testing Device

The tests have been designed to utilize the gas stabilized Arc Plasma Jet of the 40 to 100 kw size range.[i] It must produce a 1/2 inch diameter jet of air or simulated air. As seen in figures 15 and 17, a mixing chamber and 1/2 inch nozzle must be attached to the arc jet unit to produce a uniform flame. Through the use of this auxiliary nozzle, the arc may be formed on N_2, and O_2 added in the mixing chamber to reduce electrode wear.[ii] It also provides an opportunity for measuring

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[*] Dr. Robert M. Crane, Ames Research Center, Moffett Field, Calif.

[i] 40 to 100 kw size is now small. This power range is too restrictive, then, in its general application.

[ii] 1/2-inch diameter jet is too specific. Also, a mixing chamber is not required on all arcs; the arc design has progressed to beyond the state of the art when this was written.

chamber pressure.[iii] The nozzle is a straight design producing a subsonic gas stream. The initial tests will be conducted with this subsonic gas flow, although future tests will certainly include supersonic gas streams.

Test Conditions

Samples of each material system shall be exposed to three thermal environments. These were chosen to encompass the range of expected applications within the practical limitations of the equipment to be used. Therefore, the very low and the very high heat flux levels[iv] will not be attempted at this time, but will be included at a later data when more elaborate equipment becomes widely available. The test conditions are summarized in table 7.

TABLE 7.- PROPOSED STANDARD TEST CONDITIONS

Test parameter	Test number and conditions		
	I	II	III
Heat flux to cold wall calorimeter, Btu/ft^{2}sec	100	500	1,000
Time of heat pulse, sec	360	72	36
Gas velocity at nozzle[v] ft/sec approximate	2,000	2,000	2,000
Type of gas	air	air	air
Nozzle chamber pressure	Constant - Measured		
Stagnation pressure	Measured		
Power input	Constant - Measured		
Mass flow	Constant - Measured		
Gas temperature	Calculated		
Gas enthalpy	Calculated		

[iii]Establishment of subsonic nozzle as a testing standard should not be considered. The whole concept of a standardization should try to match expected environment as best as possible.

[iv]Equipment can now give low $\left(30\ \text{Btu/sec/ft}^2\right)$ as well as moderately high $\left(500\ \dfrac{\text{Btu/sec}}{\text{ft}^2}\right)$ heating rates.

[v]Gas velocity is unnecessarily low; times are unnecessarily long to establish steady-state ablation rates for these heating rates and for most ablating materials.

[redacted]

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Calibration

Accurate calibration of the arc[vi] shall be accomplished by using a water cooled copper calorimeter of the Melpar design, as shown in figures 15 and 16. This calorimeter must be used to produce correlatable data. However, information relating the values obtained using various types of calorimeters is desired and is necessary to evaluate the usefulness of the calorimeters.

The calorimeter will be used to determine the proper power setting, mass flow, etc., to produce the range of heat flux values desired. It will then be used to obtain a profile of the heat flux versus distance curve of the arc flame to determine the desired sample positions. The calorimeter shall also be used to check the heat flux at the sample location just before exposing the samples. It should be noted that the front face of the calorimeter must be polished to a bright finish before each run to obtain the proper heat transfer.

The pressure, flow rate, and temperature rise of the calorimeter water must be optimized, measured accurately, and reported with the data.

The high heat flux indicated may be unobtainable with certain facilities. In this case, the highest flux obtainable will be used and reported. Also, the power settings may be changed and reported for this test if necessary.

Standard Samples

Samples of six different, reproducible, material systems will be exposed to each of the three heat flux levels indicated. The

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[vi]The calibration of the arc should not be specified. Work needs to be done in this field. It is not generally admitted - yet true in <u>all</u> cases we have looked into - that a calorimeter gives a different (generally lower) value of enthalpy than a first law determination. Both techniques give different results from those that employ equilibrium sonic flow or direct velocity measurements. The design of the arc system seems to determine the discrepancy between the various methods. Those systems that employ the nozzle as a cathode appear to give the greatest discrepancy among the various methods. Indeed, a given arc system that gives good results at one enthalpy level may not give the same agreement at another. The causes are probably due to nonequilibrium effects. Recent heat-transfer tests performed by Griffin and Jedlicka at Ames have demonstrated that small contamination levels can change the heat transfer rate by factors as much as 3 for plasmatron arc. Therefore, it seems obvious that this method cannot be adopted as standard until more work is done in this field.

[redacted]

samples[vii] of (1) slip cast fused silica, (2) ATJ graphite, (3) Teflon, (4) Micarta, (5) 1020 steel, (6) Phenyl Silane with glass fabric. It is requested that the materials be tested in the order indicated (1 to 6) in order to quickly produce correlatable data.

The physical properties and appearance of the samples shall be recorded before and after exposure.[viii]

The sample holder should be water cooled for long life and must provide minimum contact with the sample so it doesn't affect the results by cooling, etc.

The samples[ix] shall be 1/2 inch flat plates, 2 inches by 2 inches square. They shall be exposed with a 45° angle between the flame and a face diagonal as shown in figure 18. The center of the flame shall be directed at the center of the front face.

Two spring loaded thermocouples of 28 gauge wire shall be attached to the back of the sample, as in figure 18, to record back side temperature.

Instrumentation and Data to be Obtained

The test facilities shall be instrumented to supply as much data as possible from each test. The data to be obtained and reported, along with a reasonably consistent set of units, are indicated on figure 19.

The information to be obtained on the arc shall include: power input, power loss to coolant, efficiency, mass flow rate through torch, stabilizing gas, jet gas composition, nozzle chamber pressure, gas

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[vii] It does not seem necessary to run 6 control tests. A Teflon control is desirable however because it truly sublimes and is obtainable in fairly uniform batches. The other materials would not be as good for a control because of manufacturing inconsistencies.

[viii] It is not necessary to water-cool sample holder. It is a good idea, though, to meter the heat loss from the holder and to keep it to a minimum.

[ix] The samples proposed are not good design for testing in supersonic streams. A better design would be an axially symmetric one with testing area confined to a central core area. The mass loss (of the removable central core) is then found by weighing before and after runs.

This type of model avoids edge effects and re-deposit effects for melting situations.

velocity[x] at nozzle, heat flux at sample location, stagnation pressure at sample location. Several parameters will be calculated from this data, such as: gas temperature and gas enthalpy.

The stagnation pressure shall be measured at the sample location as is shown in figure 20.

The instrumentation of the sample shall produce data such as: front side temperature versus time measured by radiation and optical pyrometers (if possible) to allow for emissivity calculations, back side temperature versus time, depth and volume of ablation or erosion, weight loss, general appearance and properties of the samples including apparent reactions and interactions, and photographs of the samples after exposure.

b. Modifications to Description of Standardized High Temperature Material Systems Screening Test[2]

Test Conditions

Due to the fact that the various classes of material systems, as typified by the "standard" samples, have greatly differing useful lives in the various thermal environments the test conditions may have been modified. These conditions, (mainly exposure time) illustrated in table 8, are designed to prevent burn through of the sample during the test, as data on thse samples would then be of questionable value.

Back face temperature measurements may be achieved through the use of the spring loaded thermocouples as was originally suggested. However, since serious error may result with this method on certain materials of low conductivity, thermocouples may be sandwiched between the back of the sample and an asbestos sheet backed up by a transite block. In any case the method used must be critically analyzed for the accuracy and the meaning of the data collected.

Calorimeter design changes[xi] as discussed at the January meeting are incorporated in figure 16. The grooved face plate should be free of stagnation areas and should provide for more uniform, higher water velocity in the critical area. As pointed out in figure 16, water inlet and outlet liner should have the same flow area, and the flow area at the face plate should be about one half of the inlet or outlet area to increase velocity.

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[x]Use velocity just ahead of model as reference standard rather than velocity at nozzle.

[xi]If the improved design of calorimeter is anything like Melpar design, it should not be used.

 [redacted]

Temperature measurement problems with thermocouples due to the electrical "noise" produced by the plasma may be circumvented by the use of battery operated potentiometers or by properly insulating the recording equipment.

TABLE 8.- PROPOSED TESTING TIME FOR VARIOUS MATERIALS

Material	Heat flux level, Btu/sq ft-sec		
	100	500	1,000
Silica	360	72	36
Graphite	120	24	12
Steel	120	24	12
Phenyl Silane	120	24	12
Micarta	120	24	12
Teflon	90	18	9

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The Nonmetallic Materials Laboratory, WADD, has been determining density of porous materials with an Amico-Winslow porosimeter. Based on the mercury intrusion technique, it is used to determine apparent density of the porous solid, and true density of the solid after reduction to particle size.[xii]

c. Testing of Reinforced Plastics Under Simulated Re-Entry Conditions[3]

The development of the plasma jet has provided a new tool for the study of the behavior of ablating systems in dissociated air. The purpose herein is to indicate the applications of the calibrated plasma jet to the experimental determination of the steady-state ablation characteristics of silica- and nylon-reinforced plastics. These materials have been selected as being typical of inorganic- and organic-reinforced plastics.

It is primarily intended to illustrate experimental techniques and possible methods of data analysis and interpretation. No attempt will be made to determine the optimum silica- or nylon-phenolic combination.

[xii]Another technique employed in industry involves the use of the Beckman air pycnometer. This instrument measures apparent density of a porous material and then with subsequent closing of the pores by means of impregnation with paraffin, a bulk density is obtained. The difference in volume is the open pores of the material. Crushing of the samples and obtaining the powder density and subtracting the apparent density will then yield the volume of the closed pores.

[redacted]

However, attention has been focused on composites in which the reinforcement constitutes the largest fraction of the mixture.

Experimental Setup

The basic experimental setup for the determination of the laminar stagnation-point ablation behavior of the reinforced plastic materials is shown in figure 21. The plasma jet was operated with a 1/2-inch subsonic exit nozzle and exhausted directly to the atmosphere. Flat-faced samples were placed on the arc axis. The sample face was located 1 inch from the exit orifice. By suitable empirical choice of exit jet and sample diameter, it was possible to obtain a match such that the sample retained its initial flat shape during ablation. In the current series of experiments,[xiii] it was determined that the 1/2-inch exit jet was best matched with 3/4-inch diameter specimens. A typical film sequence for a silica-phenolic material is shown in figure 22. It is noted that the sample retained its initial shape during the course of a run. The techniques for the determination of the thermodynamic conditions, e.g., heat flux, shear, and pressure distribution, on an "infinite" surface located 1 inch from the exit orifice, as a function of arc air flow, were summarized by John and Bade (96). It was assumed that the aerodynamic and thermodynamic conditions on the surface of the 3/4-inch diameter specimens were the same as those on an infinite plate. During the course of a typical run (30 seconds), the samples were observed to lose between 1/4- and 1/2-inch of their length due to ablation. By placing the water-calorimeter shield combination at different positions within this range along the arc axis, it was determined that there was no appreciable reduction in heat flux during an ablation test.

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Steady-State Ablation Velocity

The steady-state ablation velocity V was obtained from 35 mm Arriflex film records (24 frames/second) of the ablating sample surface. To reduce the exposure due to the radiation from the sample surface and hot plasma, the camera lens was preceded by both a Kodak Wratten No. 40 filter (peak transmission, 0.513 micron bandwidth, 0.01 micron), and a Kodak Wratten No. 2.0 neutral density filter; the camera lens was operated at an aperture of f/32. A large 12-inch focal length camera lens made it possible to obtain nearly a 1:1 object-to-image ratio.

The intense radiation from the sample surface permitted use of a slow emulsion film (Kodak Panatomic-X) with associated improved image

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[xiii]The matching of the 1/2-inch jet to a 3/4-inch model results in nonuniform heat transfer across the face. Apparently, this nonuniform heat transfer resulted in retention of flat faces for this geometry combination. In general, this matching technique is not sound because of the nonuniformities involved.

contrast and definition. Referring to figure 22, the position of the ablating surface in the individual film frames was referenced with respect to a tungsten fiducial marker. The tungsten marker and sample were silhouetted, as shown in figure 22, by means of a modified shadowgraph lighting system. It was estimated that this photographic system permitted location of the ablating sample surface to within ±2 mils.

Distance-time histories for ablating samples of silica-phenolic and nylon-phenolic at the same arc operating conditions are shown in figure 23. It is noted that the samples reached their steady-state ablation velocity in less than 5 seconds. In figure 24, the steady-state ablation mass flux (ρV) for silica- and nylon-phenolic is plotted as a function of air enthalpy. Referring to John and Bade (96) each value of the air enthalpy corresponded to unique values of the stagnation-point velocity gradient and pressure. Therefore, each value of the air enthalpy had an associated heat flux, shear, and pressure distribution. For both materials, even though the nominal aerodynamic heat input was nearly constant at about 1,250 Btu/ft^{2}sec over the operating enthalpy range, the steady-state ablation mass flux decreased with enthalpy.

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Surface Radiation

The total radiation emitted by the ablating surface and the surrounding gases was measured with a single-junction Eppley thermopile. The thermopile-mirror-ablating surface system is shown in figure 21. The geometry of the system was arranged such that the image of the flat ablating surface completely covered the thermopile junction. To estimate the total emitted radiation per unit area of sample surface, it was assumed that the angular radiation distribution was given by Lambert's cosine law. Measurements without an ablating sample in position indicated that the gas radiation was negligible.[xiv] The possibility exists of significant radiation from the boundary layer due to the presence of excited ablated species. However, the magnitude of this effect is still unresolved. Therefore for the purpose herein, it was assumed that all the measured emitted radiation originated at the ablating surface, and the radiation from the surrounding gases was negligible.

In figure 25, the radiation per unit area of the ablating sample surface has been plotted as a function of air enthalpy for the silica- and nylon-phenolic samples. For the environmental conditions characteristic of the experiments, the nylon-reinforced samples emitted nearly twice as much radiation as the silica-reinforced material. It is noted

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[xiv]There is to be considered the direction radiation from the arc. If the emissivity of the surfaces is low this radiation may be reflected to the system.

that the radiation emitted by the nylon-phenolic represented a relatively large fraction of the cold-wall aerodynamic heat input.

Surface Temperature - Spectral Determination

A lower bound for the surface temperature of ablating nylon- and silica-phenolic has been estimated by comparison of the spectral radiation characteristics of the ablating surface with that characteristic of a blackbody. The spectral distribution of the radiation from the ablating surface was measured with the monochromator (Perkin-Elmer Model 83) and mirror system shown in figure 21. The radiation was taken from the surface at an angle of about 45 degrees from the normal and was focused by a series of mirrors into the monochromator. In the monochromator, the radiation was detected by either a lead sulfide cell (0.5 to 2.5 microns) or a thermocouple detector (1 to 15 microns). The wavelength calibration was supplied by atmospheric and carbon dioxide absorption bands and by the emission lines of a mercury vapor lamp. The absolute intensity calibration was obtained by replacing the radiating sample by standard radiation reference sources.

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In figure 26, the spectral radiation emittance of nylon- and silica-phenolic is plotted as a function of wavelength in the wavelength region of from 4.1 to 5.4 microns. Superimposed on the experimental points are a series of spectral emittance plots for blackbodies at different temperatures. In this spectral region, based on measurements of the transmission of thin layers of silica, the emissivity of silica-phenolic can be expected to be high. Thus at these wavelengths of high emissivity, the blackbody curve which lies just above the experimental curve will be characteristic of the surface temperature. The surface temperature for the silica-phenolic sample is then probably close to 2,650° ± 100° K. In the case of the nylon-phenolic material, there is no strong evidence that the emissivity can be assumed close to unity. Therefore, it is only possible to conclude that the surface temperature is somewhat greater than 3,000° ± 100° K. The preceding measurements have been carried out at the environmental conditions corresponding to an air enthalpy of 7,100 Btu/lb (see table I of John and Bade)(96). Work is currently underway on the determination of the variation of surface temperature with external environmental conditions.

Surface Temperature - Melting Wire Technique

Surface temperatures of ablating specimens have also been estimated by embedding 5-mil wires of different melting points in the material sample prior to testing the sample in the plasma jet. Wire types which have been used include: iridium (mp 2,727° K), molybdenum (mp 2,890° K), and tantalum (mp 3,269° K). Methods for detection of indication of wire melting have included visual, microscopic, X-ray, and metallographic examination.

[redacted]

Definition of the Heat of Ablation

In presenting steady-state ablation results, it is found convenient to define the "heat of ablation" in a number of different forms. Four of the more common definitions are indicated below:

Cold-Wall Heat of Ablation

$q_c^* = q_c/\rho V$ (Btu/lb),

q_c = cold-wall heat input (Btu/ft^{2}sec), X650

ρ = material density (lb/ft^3), and

V = ablation velocity (ft/sec).

Hot-Wall Heat of Ablation

$$q_o^* = \frac{q_o}{\rho V} = \frac{h_s - h_w}{h_s}\frac{q_c}{\rho V} \quad \text{(Btu/lb)},$$

q_o = hot-wall heat input to nonablating surface at same surface temperature as ablating surface (Btu/ft^{2}sec),

h_s = total free-stream gas enthalpy (Btu/lb), and

h_w = total wall gas enthalpy (Btu/lb).

Radiative Heat of Ablation

$$q_{rad}^* = \frac{q_{rad}}{\rho V} \quad \text{(Btu/lb)},$$

q_{rad} = radiation from sample surface (Btu/ft^{2}sec).

Thermochemical Heat of Ablation[xv]

$$q^* = \frac{h_s - h_w}{h} + \frac{q_c}{\rho V} - \frac{q_{rad}}{\rho V} \quad \text{(Btu/lb)}.$$

The heats of ablation defined in the above equations are presented in figure 27 and figure 28 for typical high-purity silica- and

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[xv]The enthalpy term in the denominator of the thermochemical heat of ablation formula should be a free-stream gas enthalpy value. (Uncl.)

[redacted]

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nylon-reinforced phenolic (resin content 40 percent). These results serve to illustrate the necessity of clearly defining the "heat of ablation" before attempting to interpret steady-state ablation results.

Additional references for background information are noted as references 85-96.

2. LARGE ARC TECHNIQUES

INTRODUCTION

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The flight applications considered presently in the thermal protection systems field cover a broad range of environmental and material performance parameters, as well as a range of material properties and characteristics. To evaluate successfully the performance of a proposed thermal protection system, it is necessary to achieve a degree of simulation of the environment and to determine the required material characteristics under such simulated environment. Such ground tests are then translated into predicted flight performance. A number of facilities exist in the field and a number of test techniques are utilized to simulate to various degrees the anticipated flight conditions.

An effort is made here to briefly describe and review (a) the requirements for environment simulation and material performance measurement, (b) the capabilities of facilities existing in the field, and (c) operating procedures and types of test performed at Avco RAD.

The background information and detailed description employed by Avco RAD in testing are given by reference only.

REQUIREMENTS

The basic objective of any facility used to determine thermal protection system performance is to simulate the aero-thermodynamic environment in various phases of flight, and to evaluate materials proposed for the system. While a complete simulation of the trajectory and heating parameters may not be feasible at this time, some of the conditions may be reproduced.

Simulation of Environment Parameters of Interest

The significance of environmental parameters depends to a great extent on the characteristics of the materials used for thermal protection as well as flight path and applications. The necessity of transient or steady-state simula[redacted]nfluenced by the same factors and especially by the duration of the heating pulse. Generally material behavior will be affected by stagnation enthalpy (H/RT_0), heat flux (q), pressure (p), shear (τ), and duration, as well as by the pressure and shear gradient, together with the material thermal properties and ablation characteristics. The longer operating times are of particular interest in the forthcoming space applications.

The significant variables and the testing range desirable are shown in table 9 for some of the current applications. As the application range increases the requirements for parameter simulation range will increase accordingly.

TABLE 9.- ENVIRONMENTAL PARAMETERS

Vehicle	Duration of heating, $2Q_c/q_{max}$ ~ sec	Maximum heat rate range, q ~ Btu/ft^2-sec	Average stagnation enthalpy H/RT_o	Maximum local pressure range, psi	Maximum shear range, psf
Solid propellant ascent	90.0	20 - 150	20	2 - 30	4 - 12
Typical ICBM re-entry	25.0	500 - 5,000	220	10 - 230	20 - 80
	30.0	300 - 3,500	200	5 - 200	15 - 60
Satellite re-entry	200.0	10 - 110	280	0.10 - 2.0	0.40 - 12.0
Typical glide re-entry	1,800 - 4,800	10 - 120	270	Less than satellite	Less than satellite

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Measurements (Material Characteristics and Operating Conditions)

The experimental measurement required in addition to the determination of the input data (calibration) will depend on the significance of the parameters involved. The testing procedure and test requirements, including the measurement of material parameters established at Avco RAD are shown schematically in table 10 (Parts I and II). These procedures cover basically ablation and radiative types of tests from preliminary screening (Type A) to final evaluation (Test C). Additional measurements of material properties (e.g., thermal conductivity, heat capacity, chemical characteristics, etc.) are preformed as well.

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TABLE 10.- MATERIAL TESTING PROCEDURE IN PLASMA JET ARCS

Part I: Conditions (Input) and Test Requirements

	Type of test		
	Class A*	Class B**	Class C***
1. H/RT_0 - enthalpy	3 values covering the range	As defined by application if feasible, and per agreement with sponsor	As defined by application, to be selected by operator
2. τ - shear	As defined by sponsor and H/RT_0		
3. p - pressure			
4. $\dot{q}$ - heat flux			
5. t - running time			
6. n - number of samples	6	18	Depending on 1 through 5 requirements
7. ρ - density	Per sponsor information	Per sponsor information	Per sponsor information
8. ϵ - emissivity			
9. T_s - surface temp.			
10. Composition			
11. Calibration (operating conditions)	General	Specific (3 runs) over the range if necessary	Same as B but more frequent
12. Test facility (pipe, shroud, etc.)	Pipe and splash	Pipe	Pipe, shroud, HTT, etc., selected by operator depending on application
13. Application	Per sponsor	Per sponsor	Per sponsor

*Preliminary.
**Intermediate.
***Final qualification.

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TABLE 10.- MATERIAL TESTING PROCEDURE IN PLASMA JET ARCS - Concluded

Part II: Results and Report Requirements

	Type of test		
	Class A*	Class B**	Class C***
14. q^*	Yes - vs. H/RT_o	Yes - vs. H/RT_o	Vs. H/RT_o
15. η - transpiration factor	Yes - graphical	Yes - graphical	Yes - curve fit as applicable
16. H_v - heat of ablation	Yes - graphical	Yes - graphical	Yes - curve fit
17. f - evaporation fraction or efficiency factor	No	Yes - graphical	Yes - curve fit and measurement
18. M - mass loss	Total	Total	Total and rate
19. ΔL - thickness loss	Total average	Total average and traverse	Total and rate and traverse should include film coverage
20. Visual observation: (a) appearance (b) brightness (c) blinking, etc. (d) flow, etc. (e) other	 Yes Yes Yes Yes Yes (per sponsor)	 Yes Yes Yes Yes Yes (per sponsor)	 Yes Yes Yes Yes Yes
21. Basic input data (see 1 - 13 in Part I)	Yes	Yes	Yes
22. Estimate of accuracy	General input only	Specific - input and output	Specific - input and output statistical evaluation
23. Temperature dist.	No	Yes - 3 runs spot check	Detailed traverse
24. Description mechanism of process	No	No	Yes - include theoretical background, description by reference if available
25. Analytical comparison	No	No	Yes
26. Char analysis section	No	No	Yes
27. Report form by operator	TRR	TRR	TM or TR

*Preliminary.

**Intermediate.

***Final qualification.

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A. The 500 KW Arc Facility for Splash Tests (Avco "500" Arc)

The arc-jet unit used in the experiments is shown schematically in figure 36. The gas to be heated is injected tangentially into the chamber. The arc discharge is ignited by exploding a tungsten bridge wire which initially connects the carbon cathode and water-cooled copper anode. The injected gas flows through the region occupied by the arc discharge, momentarily becomes part of the discharge, and then leaves the system through a plenum chamber and a subsonic exit nozzle. The entire arc chamber, including the anode, plenum chamber, exit nozzle, and cathode holder, is water-cooled.

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The basic experimental arrangement used for determining the laminar stagnation-point ablation behavior is shown in figure 29. The arc-jet unit may be operated with a 1/2 inch subsonic exit nozzle, and the jet was exhausted directly to the atmosphere. Flat-faced samples are placed on the jet axis. The face of each sample is located one inch from the exit orifice. By suitable empirical choice of jet and sample diameters, it is possible to obtain a match such that the sample retained its initial flat shape during ablation. The 1/2-inch plasma jet was best matched with 3/4-inch diameter specimens.

The calibration of an arc-jet unit consists of the determination of the aerodynamic and thermodynamic parameters which characterize the interaction of the subsonic jet of arc-heated air with the surface of a flat-faced material specimen. The nominal heat flux, shear, and pressure distributions on an "infinite" flat surface located one inch from the exit orifice are summarized in table 11 as functions of jet mass flow for conditions of "low" and "high" arc input power.* In the splash technique of materials testing, it is assumed that the inch diameter specimens are the same as those on an infinite plate.

Detailed descriptions of Avco RAD facilities, testing procedures and background information covering theory of operation and of the ablation mechanism used by Avco in the interpretation of test results are given in the reference list attached. The importance of proper experiment design and interpretation and comparison with the theory cannot be overemphasized.

B. The 10 Megawatt Arc Complex

The 10 megawatt air stabilized arc consists of 5 individual arcs discharging into a common plenum. Of these 5 arcs up to 4 can be operated simultaneously for a total power input of 10 mw. This multiple arc can be operated with mass flow rates from 0.02 to 0.5 pounds

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*The designations "low" and "high" input power refer to two settings of the d-c power source used to supply the plasma jet.

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TABLE 11.- STAGNATION-POINT AEROTHERMODYNAMIC CONDITIONS

CHARACTERISTIC OF THE PLASMA JET

(a) Low input power (0.12-ohm ballast)

Air mass flow (lb/sec)	Air enthalpy (Btu/lb)	Stagnation pressure (atm)	Velocity gradient (sec^{-1})	Heat flux (Btu/ft^2sec)	Shear-stress gradient (lb/ft^3)	Second derivative of pressure (lb/ft^4)
0.006	8,600	1.06	48,000	930	160	-0.18×10^6
.010	6,300	1.14	61,000	930	280	-.38
.014	5,000	1.25	72,000	930	420	-.68
.018	4,100	1.37	79,000	930	560	-1.04
.022	3,500	1.51	85,000	930	700	-1.48

(b) High input power (0.06-ohm ballast)

Air mass flow (lb/sec)	Air enthalpy (Btu/lb)	Stagnation pressure (atm)	Velocity gradient (sec^{-1})	Heat flux (Btu/ft^2sec)	Shear-stress gradient (lb/ft^3)	Second derivative of pressure (lb/ft^4)
0.006	10,400	1.07	53,000	1,200	190	-0.19×10^6
.010	8,300	1.18	74,000	1,200	330	-.48
.014	6,800	1.32	89,000	1,300	530	-.89
.018	5,800	1.49	100,000	1,300	700	-1.41
.022	5,100	1.67	109,000	1,300	860	-2.05

of air per second, and stagnation plenum chamber pressures from 0.5 to 26 atmospheres absolute. Precise limits of operation are listed in table 12.

The high available power permits large air mass flows to be heated to re-entry enthalpies, thus opening up a wide range of possible aero-thermodynamic experimental facilities, four types of which are presently in operation at Avco: splash, pipe, shroud, and hyperthermal wind tunnel.

1. Splash Technique

In a subsonic splash facility air flows from the settling chamber through a subsonic nozzle into a pressurized atmosphere. The jet impinges upon and flows normal to a blunt cylinder, simulating the stagnation point conditions. The operating capabilities presented as the variation of stagnation point heat transfer rate with pressure for a typical splash configuration is shown in figure 29. Lines of constant throat diameter are cross plotted. The effect of variation in exit Mach number at a fixed stagnation enthalpy is also shown in the figure.

With the splash technique considerable test flexibility is possible because, for a constant throat diameter and throat Mach number the power and enthalpy can be varied. Or for a fixed jet exit diameter the enthalpy can be fixed and the power and exit Mach number varied.

2. Pipe Technique

This technique is very successfully used for turbulent ablation studies. The arc heated air flows from the stagnation plenum through a pipe at the end of which is a sonic throat for flow control. Ablation is measured in the pipe region after boundary layer transition but before fully developed pipe flow is obtained. Typical capabilities for turbulent boundary pipe flow are presented in figure 30. The heat transfer rates based on axial distance are shown for fixed enthalpy as a function of non-dimensional length. The values of Reynold's number and non-dimensional boundary layer are shown cross-plotted in the figure.

3. Shroud Technique

In a subsonic shroud facility, air flows from a plenum chamber through an annular region formed between the shroud and the model. The shroud is contoured with respect to the model to obtain a prescribed pressure distribution.

The shroud technique which has been used for blunt body heat transfer study has now been adapted to ablation research. The inset sketch in figure 31 illustrates the arrangement. The approximate capabilities

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TABLE 12.- CHARACTERISTICS OF AVCO PLASMA ARC FACILITY

1. Type of test	10 megawatt facility: Shroud	10 megawatt facility: Wind tunnel	10 megawatt facility: Pipe	10 megawatt facility: Splash	High enthalpy arc: PG-500 plasma generator	Low enthalpy arc: PG-500 plasma generator (modified)
2. Enthalpy level attainable						
(a) Min. (H_s/RT_o)	8	8	8	8	100	10
(b) Max. (H_s/RT_o)	350	350	350	350	300	100
3. Power (megawatts)					Arc 0.178, arc 0.060	Arc 0.150, air 0.035
(a) Input (max.)	10	10	10	10		
(b) Output (max.)	3.5	3.5	3.5	3.5		
4. Chamber pressure (atm.)						
(a) Min. enthalpy level	2.5 to 26	0.5 to 26	0.5 to 20	2 to 26	1.41	1 to 1.7
(b) Max. enthalpy level	2.5 to 4	1 to 9	1 to 9	2 to 9	1.07	1 to 1.7
5. Stagnation pressure (atm.)						
(a) Min. enthalpy level	2.5 to 26	0.02 to 1.5	----------	1 to 13	1.41	1 to 1.7
(b) Max. enthalpy level	2.5 to 4	0.04 to 0.5	----------	1 to 4.5	1.07	1 to 1.7
6. Static pressure (atm.)						
(a) Min. enthalpy level	1 to 10	7×10^{-4} to 3.6×10^{-2}	0.3 to 12	1	1	1
(b) Max. enthalpy level	1 to 1.6	1.4×10^{-3} to 14×10^{-3}	0.6 to 5	1	1	1
7. Mass flow (lb/sec)						
(a) Min. enthalpy level	0.7 to 5	0.08 to 4	0.15 to 3	0.5 to 3	0.0182	0.00635 to 0.0508
(b) Max. enthalpy level	0.18 to 0.3	0.03 to 3	0.02 to 0.4	0.1 to 0.35	0.0056	
8. Mach number capability	----------	4.5 to 7	0.6	0.8 to 1.5	0.3 to 0.8	0.3 to 0.8
9. Model size (max.)	6 in. diam. Hemisphere	3.5 in. diam.	1.5 in. diam.	3 in. diam.	0.750 in. right circular cylinder 0.750 in. right circular cylinder (solid)	
10. $\dot{Q}_s$ (cold wall) (Btu/ft^2-sec)						
(a) Min. enthalpy level	----------	Hemisphere 2 to 10	To 100	To 100	800	20 to 250
(b) Max. enthalpy level	1,700 to 2,400	100 to 400	200 to 1,500	200 to 1,500	1,200	20 to 250
11. Test time (secs.)	60	60	60	60	300 (max.) 30 (nom.)	Up to 300
12. Reynolds no./in.						
(a) Min. enthalpy level	----------	3×10^4 to 4×10^5	5×10^5 to 1×10^7	----------	N/A	N/A
(b) Max. enthalpy level	*4×10^4 to 7×10^4	1×10^2 to 5×10^3	5×10^4 to 2×10^5	To 3×10^5	N/A	N/A
13. Contamination level versus enthalpy (percent of air)	0.5 to 4	0.5 to 4	0.5 to 4	0.5 to 4	4.8 (max.) 1.2 (min.)	5
14. Shear level versus enthalpy (lb/ft^2)		Function of model	2 to 32	Function of model	N/A	N/A
15. Composition of plasma (at model) and degree of equilibrium established	Not determined				Various gases may be run in the arc-air, nitrogen, argon, helium, etc.	Different gases have been used as the working fluid air oxygen, nitrogen, argon, and helium.
16. Method of calibration for $\dot{Q}_s$ (cold wall)	Water calorimeter	Copper plug with T.C.	Water and heat sink	Water calorimeter and plug with T.C.	Cold wall heat flux, q_{cw}, is determined by placing a water cooled copper calorimeter of identical size as the specimens tested, in the same position as the samples tested. The flow rate and temperature rise of the water through this calorimeter are recorded, and q_{cw} is then calculated. Radiation from the calorimeter is negligible. Radiation from tested specimens is recorded using a total radiation thermopile.	Cold wall heat flux (q_c) is measured with water-cooled calorimeter that has the same surface area as the ablating sample. The sides of the calorimeter are protected by a water-cooled shield. The flow rate and temperature rise of the cooling water give the heat transfer.

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of the shroud and the operating limits of the arc are shown in figure 31. This figure presents cold wall stagnation heat transfer, calculated from Fay-Riddell theory, to a typical six-inch diameter hemispherical shroud model as a function of stagnation pressure for various dimensionless enthalpy ratios. Lines of constant power input to the air and air mass flow are shown. Power to the air is simply equal to the mass flow times the enthalpy. The maximum power that can presently be imparted to the air is about 3-1/2 megawatts (3,300 Btu/sec) thus forming one limit on the test envelope. The other limits are a maximum enthalpy ratio of about 300 (10,000 Btu/lb), maximum total pressure of 20 atmospheres
X although pressures up to 26 atmospheres have been reached; and for
6 exhausting to the atmosphere, a minimum pressure of about 1.8 atmos-
5 pheres to insure chocking at the sonic point. Other larger and smaller
0 shrouded models can also be adapted to the arc.

It is known that true simulation of aerothermodynamic hypervelocity flight requires simulation of enthalpy, heat flux, and to a lesser extent pressure particularly where ablating designs are under study. The typical ICBM trajectory shown on figure 31 shows heat flux - pressure simulation. Since the model size is similar to a full scale re-entry vehicle nose, the enthalpy is also very nearly simulated. Thus, a high degree of re-entry simulation with the shroud has been demonstrated.

4. Hyperthermal Wind Tunnel

This is a supersonic wind tunnel which enables the determination of ablation characteristics of scale models. Such facility called the hyperthermal wind tunnel and capable of direct simulation of re-entry conditions for relatively long test durations is now in operation.

This facility is a blow-down nominal Mach 5 very high temperature wind tunnel with a one-foot diameter test section. A cutaway drawing of the tunnel is shown in figure 32. Figure 33 shows a photograph of the facility with a super-satellite re-entry model mounted in the test section. Models up to 4 inch diameter can be tested for duration of up to one minute.

Simulation Capability

The aerothermodynamic environment simulation capability of the tunnel is shown in figure 34. Stagnation heat transfer on a 3-1/2 inch diameter Mercury shaped model is shown as a function of stagnation pressure for various enthalpies. The various facility limits indicated on the figure form a usable test envelope for H_S/RT_O of 7.5 up to perhaps 800, and model stagnation pressures from 0.002 up to 0.5 atmospheres.

A typical supersatellite re-entry overshoot trajectory for a 10 foot diameter Mercury shape is crossplotted on this figure in two forms;

stagnation heat transfer versus stagnation pressure, and versus the skewed H_s/RT_0 grid. From these two curves the high degree of re-entry heating simulation obtainable is evident. It follows that simulation of flow conditions around the model is also achieved.

Figure 35 shows the test section conditions obtainable for total pressure of 2 and 20 atmospheres including density, density altitude, Reynolds number, total and static temperature, as functions of velocity and total enthalpy for equilibrium flow. Since the flow in the nozzle departs somewhat from equilibrium, the actual velocities and temperatures are lower and the Reynolds numbers higher than shown.

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Test Methods

Pressure distributions on either cooled or uncooled models can be measured directly with conventional pressure transducers. Heat flux measurements can be made with copper plug calorimeters which utilize imbedded thermocouples. Similarly, temperature distributions within and around the model can be recorded using conventional thermocouple methods.

The design of a one-component sting mounted water cooled balance for direct measurement of drag forces is available, and for simultaneous measurement of lift and drag forces an external balance is being considered.

The tunnel is unique in its ability to reproduce ablation conditions over significant time lengths with complete flow field simulation. Thus the complete aero-thermodynamic performance of a vehicle design in the laminar flow region can be studied in one facility.

A Schlieren and shadowgraph flow visualization system is in operation permitting the study of flow fields, shock shapes, etc. Furthermore, body shape as a function of time for ablation tests can be photographed directly by fast motion pictures or indirectly through the Schlieren system. Measurement of surface radiation and/or surface temperature can be accomplished by spectroscopic techniques.

Projected Improvements

Within a few months* the hyperthermal wind tunnel will be capable of virtually continuous operation which will permit tests to be conducted with long heat pulses. Likewise a Mach number 12 nozzle is under design for testing at simulated altitudes above 300,000 feet.

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*As of June 1961.

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3. REVIEW OF FACILITIES

A review of existing facilities suitable for partial environment simulation was made and a summary of their capabilities is given in table 13. In many cases only an estimate of the capability was made because of difficulties in obtaining properly referenced information. A brief definition of terminology used is included in the table and the range of operating parameters is given.

Avco-RAD facilities used for simulation of the environment are
X described more in detail below. Testing procedures, types of test and
6 test techniques are given, and a detailed summary of operating condi-
5 tions (including the range of parameters attainable) are shown in
0 table 12.

An added review of high temperature evaluation information is given in table 14, taken from reference 86, WADD TR60-90.

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TABLE 13.- SOME FACILITIES FOR TESTING THERMAL PROTECTION SYSTEMS AND MATERIALS PERFORMANCE*

Name of facility		Chance Vought	Plasmadyne	General Electric	AVCO	AVCO	Cornell Aeronautical Laboratory	Chicago Midway Laboratory
Location of facility		Dallas, Texas	Santa Ana, Calif.	Philadelphia, Penn.	Wilmington, Mass.	Wilmington, Mass.	Cornell University, New York	University of Chicago
Test apparatus		Ramjet heater	Air plasma arc water cooled electrodes	Arc heating supersonic wind tunnel	10 megawatt arc (1) Shroud (2) Wind tunnel (3) Pipe (4) Splash	Overs II arc	High temperature resistance heater N_2	Plasma air arc
Source of information		Private communication with Chance Vought to AVCO	Estimated	Material for re-entry heat protection of satellites	Internal company communications	Internal company communications	WADC TR-368 Part II, study of ablation of structural plastic materials	WADC TR-59-87, determination of factors governing selection and application of materials . . .
Operating conditions	Run duration	Up to 20 minutes	Up to 20 hours	2 minutes	100 seconds**	Up to 60 minutes	Up to 60 seconds	60 seconds
	Power supply	Ramjet burning JP-4 fuel at combustion pressure 50 psi	100 to 1,000 kw using rectifiers		8 megawatts	100 to 1,000 kw	1,600 kw	1,000 kw
	Power output		50 to 700 kw to gas		3.5 megawatts to gas	50 to 500 kw	65 kw	
	Heat flux	127 Btu/ft^2-sec to cold wall at full power 80 in. from exhaust plane to a cold water calorimeter	2,000 Btu/ft^2-sec to $4\frac{5}{8}$ in. dia. calorimeter	65 Btu/ft^2-sec	(1) Up to 2,400 Btu/ft^2-sec (2) Up to 400 Btu/ft^2-sec (3) Up to 1,500 Btu/ft^2-sec (4) Up to 1,500 Btu/ft^2-sec	0.1 to 3 kw/cm^2	700 Btu/ft^2-sec (max.)	2,000 Btu/ft^2-sec cold wall
	Flow rate	Max. 20 lb/sec with 1.4 lb/sec fuel flow rate; min. 8 lb/sec	$\left.\begin{matrix}\text{Max. 0.10 lb/sec}\\\text{Min. 0.01 lb/sec}\end{matrix}\right\}$ M = 2 to 3	M = 4.8	$\left.\begin{matrix}(1)\ 0.18\text{ to }5\\(2)\ 0.03\text{ to }4\\(3)\ 0.02\text{ to }3\\(4)\ 0.10\text{ to }3\end{matrix}\right\}$ lb/sec	0.003 to 0.1 lb/sec	40 lb/ft^2-sec (max.)	0.25 lb/sec
	Enthalpy	Calculated to be* 18,000 Btu/lb; not yet verified experimentally	200 to 25,000 Btu/lb	5,100 Btu/lb	200 to 12,000 Btu/lb	2,000 to 25,000 Btu/lb		
	Gas temperature	Up to 4,000° F at stagnation theoretical value	3,000° K to 8,500° K			3,000° K to 10,000° K	4,000° F	4,800° F (stag.)
	Pressure in chamber	Up to 50 psi in stagnation diam.	0.10 to 40.0 atmospheres in arc chamber		$\left.\begin{matrix}(1)\ 2.5\text{ to }26\\(2)\ 0.5\text{ to }26\\(3)\ 0.5\text{ to }20\\(4)\ 2.0\text{ to }26\end{matrix}\right\}$ atmospheres	0.03 to 5 atmospheres		
	Stagnation pressure				$\left.\begin{matrix}(1)\ 2.5\text{ to }26\\(2)\ 0.02\text{ to }1.5\\(3)\\(4)\ 1.0\text{ to }13\end{matrix}\right\}$ atmospheres	0.005 to 5 atmospheres		
Instrumentation		Thermocouples, pressure gages, flow calibration	Flow calibration calorimeter pressure probes enthalpy probes velocity devices radiation devices spectroscopic measurements	Thermocouples calorimeter	Thermocouples calorimeter		Water cooled calorimeter radiation pyrometer thermocouples	Copper calorimeter fusible calorimeter spectroscope color motion pictures
Sample geometry		$1\frac{1}{2}$ in. to 15 in. diameter hemispheres at various distance from exhaust plane	Cones, cylinders, hemispheres	Blunt nose cylinder	(1) 6 in. diameter (max.) (2) 3.5 in. diameter (3) 1.5 in. diameter (4) 3.0 in. diameter	6 in. diameter standard 3 in. to 8 in. nozzle exits available	Cylinder	Spheres cones

*Data taken from reports gives facility characteristics for the particular test reported and not necessarily the entire operating range.
**Being extended.

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TABLE 13.- SOME FACILITIES FOR TESTING THERMAL PROTECTION SYSTEMS AND MATERIALS PERFORMANCE* - Concluded

Name of facility		Stanford Research Institute	Wright Air Development Division	Thermodynamics Laboratory	Aerojet General Corp.	Langley Research Center	Ames Research Center	AVCO
Location of facility		Stanford University	Dayton, Ohio	Convair San Diego, Calif.	Azusa, Calif.	Langley Field, Va.	Moffett Field, Calif.	Wilmington, Mass.
Test apparatus		(1) Arc image: air and argon (2) Argon plasma arc	(1) Plasma arc: air or argon (2) Supersonic rocket	Hydrogen rocket motor	Rocket motor	10 mw arc tunnel	Arc-jet, arc image furnace	Model 500 arc
Source of information		WADC TR 59-668 Part II, study of the mechanism of ablation of reinforced plastic	WADD, studies on plastic exposed to high mass flow thermal environments	WADD, effects of high temperature high velocity gases on plastic materials	WADD TR 60-193, thermal erosion of ablative materials	Internal communications	Internal communications	Internal company communications
Operating conditions	Run duration	(1) 40 seconds (2)	(1) 30 seconds (2) 5 seconds	12 seconds	20 seconds	60 seconds	12 seconds continuous	5 minutes
	Power supply	(1) (2)	(1) (2) 41 kw			10 mw AC	3.5 mw max. DC generators	35 to 150 kw
	Power output		(1) (2)			-------------------	100 to 600 kw (input)	8 to 85 kw
	Heat flux	(1) 500 Btu/ft^2-sec (2) 750 Btu/ft^2-sec	(1) (2) 1,500 Btu/ft^2-sec	3,550 Btu/ft^2-sec	1,150 Btu/ft^2-sec		0 to 200 Btu/ft^2-sec	20 to 250 Btu/ft^2-sec
	Flow rate		(1) (2) 0.031 lb/sec	Mach no. = 1.82 (max.)		0.017 to 1.0 lb/sec	0.0005 to 0.10 lb/sec	5 to 40 ft^3/min
	Enthalpy		(1) 3,300 Btu/lb (2) 2,600 Btu/lb	3,600 Btu/lb		8,000 Btu/lb	1,700 to 11,000 Btu/lb	10 to 100 Btu/lb
	Gas temperature		(1) 6,300° R (2) 3,740° R	6,330° F	5,500° F	To 16,000° R		
	Pressure in chamber					600 to 9,400 psia in settling chamber	300 microns to 2 atmospheres	0 to 14.7 psig
	Stagnation pressure				500 psia	0.09 to 1.4 psia	0.25 to 3 atmospheres	1 to 2 atmospheres
Instrumentation					(1) Continuous chamber pressure recording	Digital data processing calorimeters spectrography motion pictures temperature measurement pressure measurement	Pressure, temperature, and rate of recession on oscillographs and film	(1) Water cooled copper calorimeter (2) Thermocouple in sample (3) Pyrometer surface temperature (4) Thermopile total radiation (5) Camera front surface and side
Sample geometry		Hemispheres cones		Cylindrical cone	Nozzle inserts	9 in. diam. streamlined bodies	Nose radius equals 2 times cylinder diam.	Cylinders spheres

*Data taken from reports gives facility characteristics for the particular test reported and not necessarily the entire operating range.

[redacted]

TABLE 14.- HIGH TEMPERATURE EVALUATION FACILITIES

Type of facility	Gas enthalpy (Btu/lb)	Gas velocity (fps)	Gas mass-flow (lb/sec)	Stagnation point conditions		Test fluid composition	Available testing time (min)	Test area diameter (ft)	Principal advantages	Principal disadvantages
				Pressure (psf)	Initial heat flux (Btu/ft^2-sec)					
Chemical torch (oxy-acetylene)	1,000	100 to 750	0.002 to 0.021	50 to 1,000	100 to 650	Combustion products	Near continuous	0.04 to 0.14	Inexpensive. Simple and rapid materials screening device.	Low enthalpy. Small exposure area. Chemical reactivity of gases.
Rocket motor (oxy-gasoline)	1,500 to 3,500	7,000 to 8,000	0.5	10,000 to 80,000	300 to 3,000	Combustion products	0.5 to 10	0.08 to 0.70	Simulates actual rocket exhaust environment. Accommodates large model sizes. High stagnation pressures and supersonic flow.	Inability to change heat flux without greatly affecting stagnation pressure.
Electric arc heater (500 kw)	1,000 to 12,500	1,500 to 2,500	0.005 to 0.040	2,000 to 4,000	100 to 1,500	Air, argon, nitrogen, or helium in molecular dissociated and ionized states.	Near continuous	0.03 to 0.10	Means for continuously heating variety of gases to very high temperatures.	Low mass-flow. Small exposure area. 0.1 to 5 percent electrode contamination of gas. High power requirement.
Electric air arc tunnel (500 kw)	2,500 to 12,500	2,000 to 15,000	0.005 to 0.040	200 to 2,000	100 to 1,500	Air in molecular, dissociated and ionized states.	Near continuous	0.03 to 0.15	Re-entry simulation.	Low mass-flow. Small exposure area. 0.1 to 5 percent electrode contamination of gas. High power requirement.
Graphite-resistor furnace	1,300	3,000	0.03	6,000	200 to 800	Inert gases or air	0.1 to 3.0	0.04 to 0.10	Inexpensive. Versatile.	Erosion of furnace walls. Small exposure area.
Pebble bed heater	1,200	3,600 to 5,500	10	70 to 15,000	50 to 400	Air	0.1 to 25	0.04 to 0.08	High stagnation pressure.	Small exposure area. Air temperature declines (10° F/sec) with exposure time. 0.5 to 1.0 percent ceramic dust contamination of gas.
Arc image furnace	--------	0	0	0 to 2,000	50 to 2,500	Vacuum, or any gas	15 to 30	0.02 to 0.08	Non-contaminated heat source. Control of environmental conditions. Variety of available chemical environments.	No imposing aerodynamic flow. Highly absorbent surface required. Small exposure area. Surface volatiles interfere with radiant transmission.

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APPENDIX B

NOMENCLATURE AND SYMBOLS

Proposed standardized definitions, terms, symbols, and dimensions are as follows:

a. General terms

alt	altitude, mi or ft
v	velocity, ft/sec
U or u	re-entry velocity, ft/sec
t	time, sec
M	Mach number
d	diameter, in
r	radius (nose, etc.), in

b. Material constants

k	thermal conductivity
ϵ	emissivity (total hemispherical)
α/ϵ	ratio of absorptivity to sunlight and IR emissivity
ρ	density, lb/in^3
c_s	specific heat solid state, Btu/lb/$^{\circ}$F
c_l	specific heat liquid state, Btu/lb/$^{\circ}$F
H_f	heat of fusion, Btu/lb
H_v	heat of vaporization, Btu/lb

c. Environmental terms

$\dot{m}$	mass flow (environmental gas), lbs/ft^2/sec
H	gas enthalpy, Btu/lb

[redacted]

H_W gas enthalpy at the wall

R_e Reynolds number

γ ratio of specific heats c_p/c_v

p pressure

ρ density

V total volume of gas, lbs X650

μ viscosity of gas, lbs/sec/ft

c_p, c_v specific heat, Btu/lb/^{o}F

Environmental subscripts

s stagnation point

t total

W wall

∞ free stream

a ambient

d. Heat exchange

Q total heat input, Btu/ft^2

heat transfer coefficient

$\dot{q}$ total environmental heat flux to surface (convective and radiant), Btu/ft^2/sec

$\dot{q}_{max}$ representative environmental heat flux to surface (fig. 2)

$\bar{t}$ representative time of heat exposure, depending on TPS according to figure 2

$\dot{q}_c$ convective heat flux to surface for cold wall, Btu/ft^2/sec

$\dot{q}_h$ reduction of $\dot{q}_c$ or q_{max} for hot surface

[redacted]

$\dot{q}_b$ — reduction of net heat flux to the surface for "blocking action" (or blowing effect), $\dot{q}_b = \eta \dot{m}(H - H_w)$

η — constant depending upon the coolant

$\dot{q}_R$ — radiant heat flux from gases to surface

$\dot{q}_r$ — heat reflected at hot surface

$\dot{q}_e$ — heat flux emitted from a hot surface, $\dot{q}_e = \sigma\epsilon\left[T_{abs}\right]^4$, Btu/ft^2/sec

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$\dot{q}_a$ — heat flux absorbed in active part of TPS

$\dot{q}_s$ — heat absorbed in base material and/or support structure

σ — Stefan-Boltzmann constant (0.1713×10^{-8}), Btu/ft^2/sec/^{o}F^4

e. Material capabilities

Q^* — effective thermal capacity for a certain heat flux range to be specified by subscripts, $Q^* = \dot{q}/\dot{m}$, Btu/lb

Subscripts

a	ablation	$Q^*_{a\ \mathrm{or}\ t} = \dfrac{\dot{q}_c - \dot{q}_h - \dot{q}_e - \dot{q}_b}{\dot{m}}$
t	transpiration	
s	non-melting heat sink	$Q^*_{s\ \mathrm{or}\ c} = \dfrac{\dot{q}_c - \dot{q}_h - \dot{q}_e}{\dot{m}}$
c	convective cooling and/or radiative	
f	total capacity to include melting	
v	total capacity to include vaporization or sublimation	

W_t — effective total weight of TPS and support structure (fig. 5), lb/ft^2

W_n — effective net weight of TPS (active portion), lb/ft^2

Δw — effective weight of passive portion of TPS (back-up weight, pumping systems), lb/ft^2

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[redacted]

w_g	effective gross weight = $W_n + \Delta w$
$\dot{a}$	ablation rate (volume), in/sec
a	total ablation (volume), in
$\dot{W}$	ablation rate (weight) or coolant flow, $lb/ft^2/sec$

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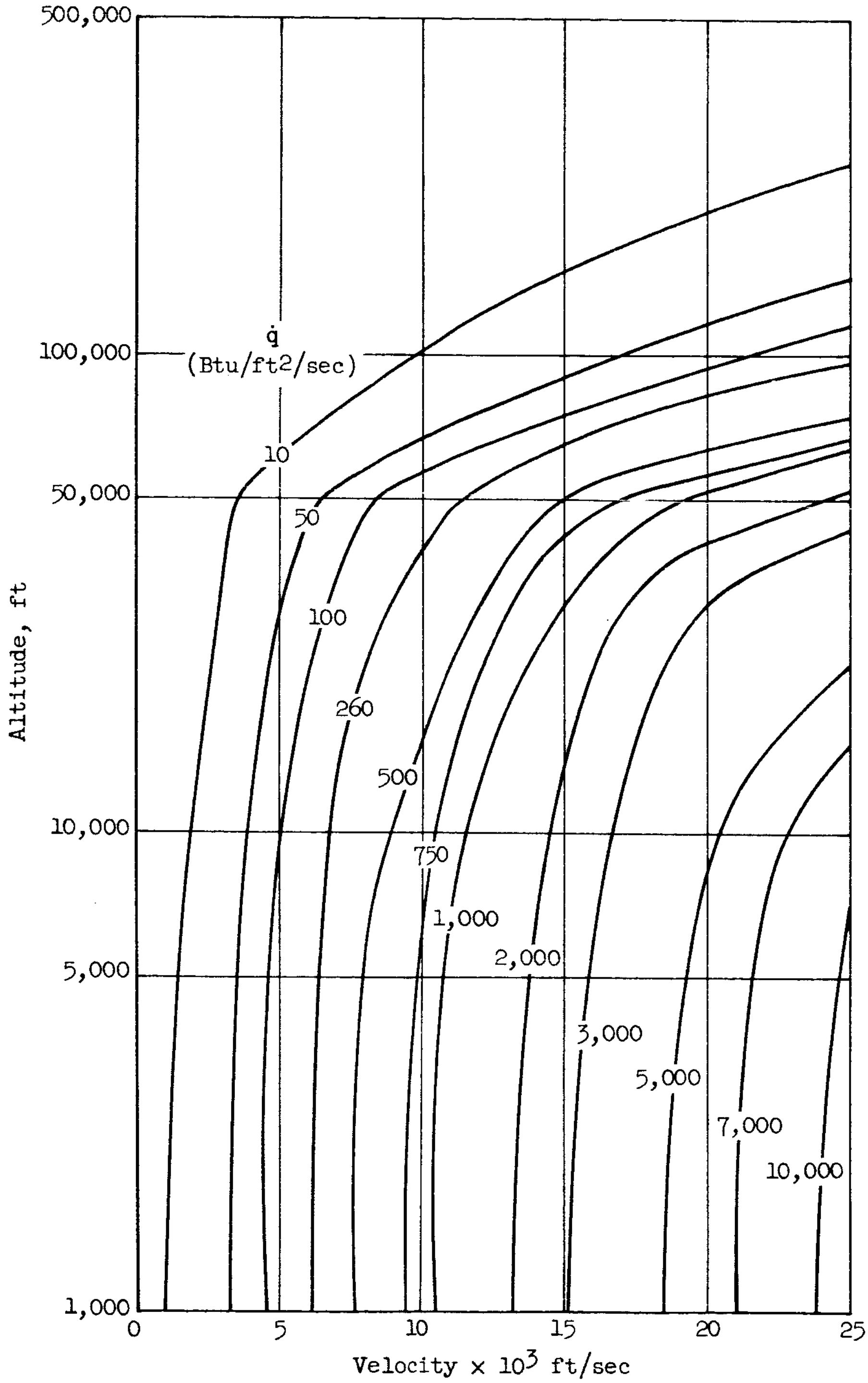

Figure 1.- Typical stagnation-point convective heat flux in relation to velocity and altitude.

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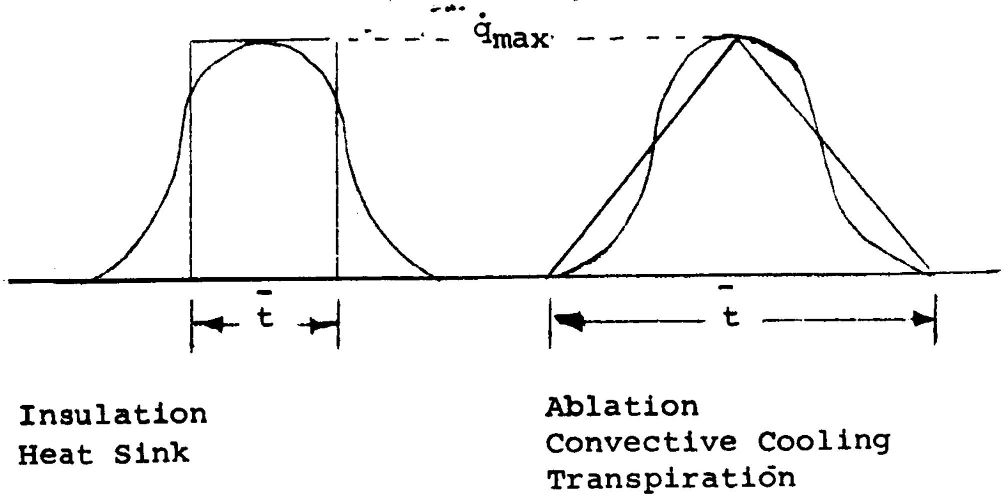

Figure 2.- Representation of the transient heating cycle.

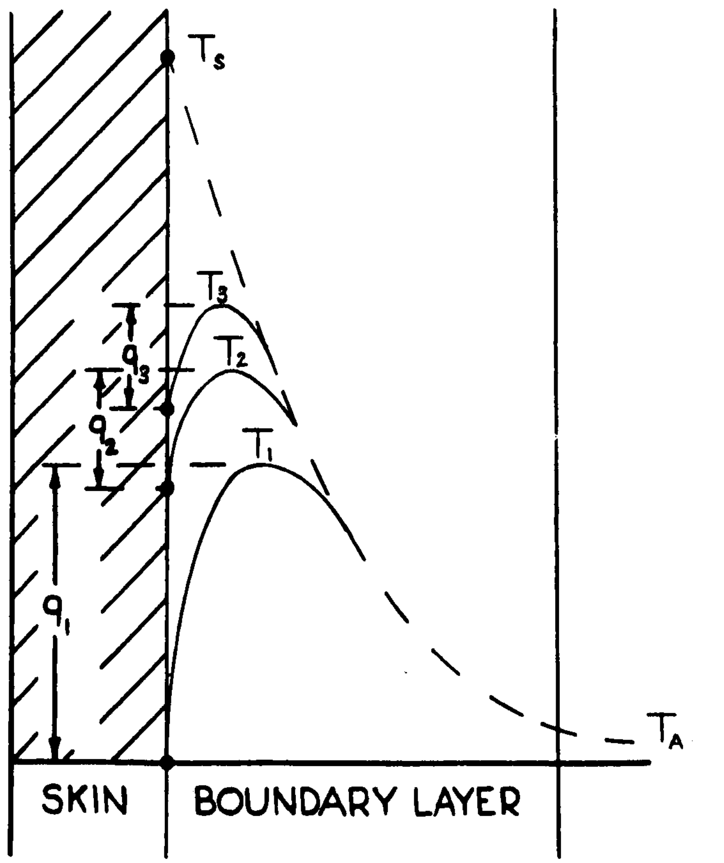

Figure 3.- Changing heat flux and BL temperature profile during a transient heating cycle.

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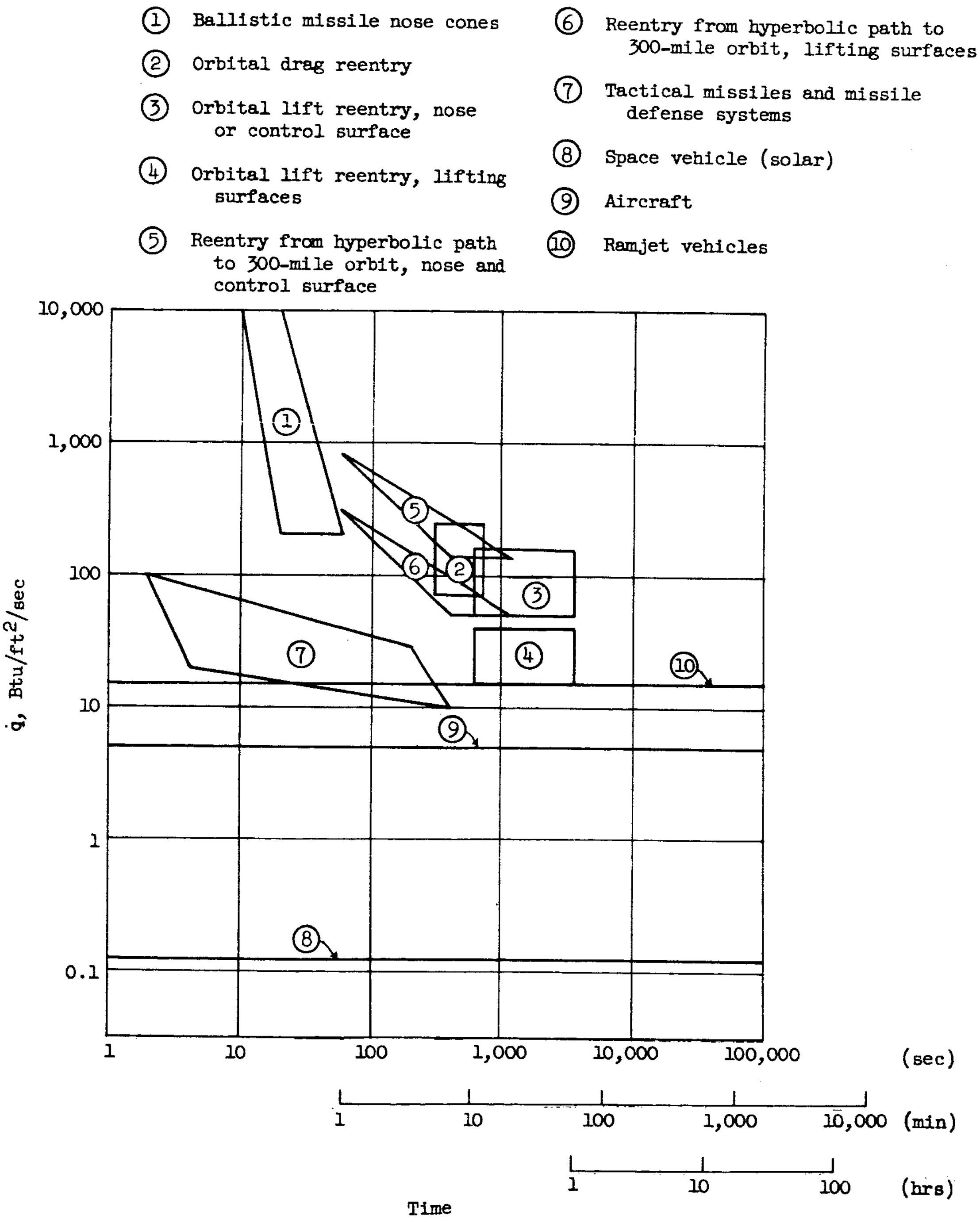

Figure 4.- Thermal environments of advanced vehicle systems.

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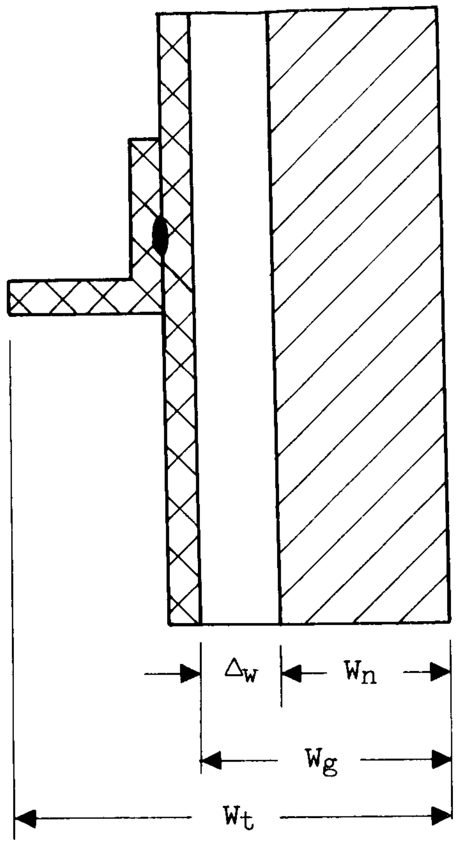

Figure 5.- Weight designations.

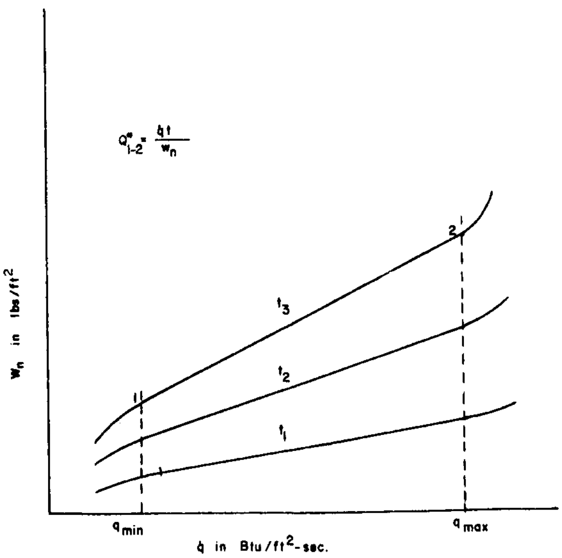

Figure 6.- Experimental determination of Q^*.

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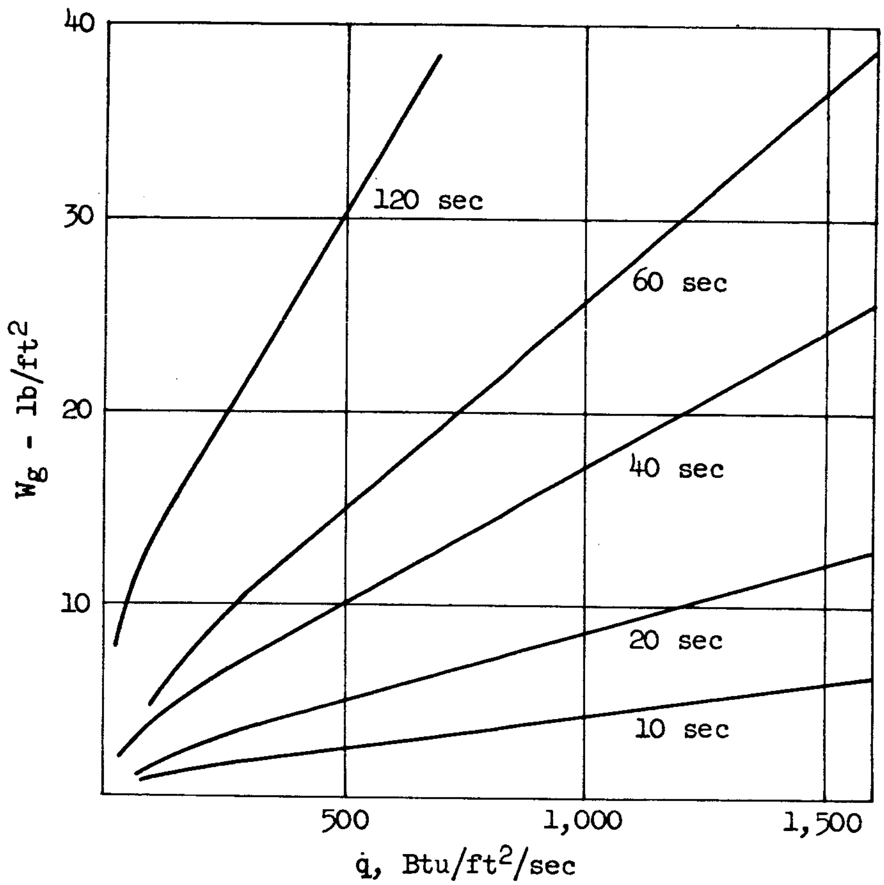

Figure 7.- Typical gross-weight-heat-flux chart for an ablation material (77% glass fiber reinforced phenolic).

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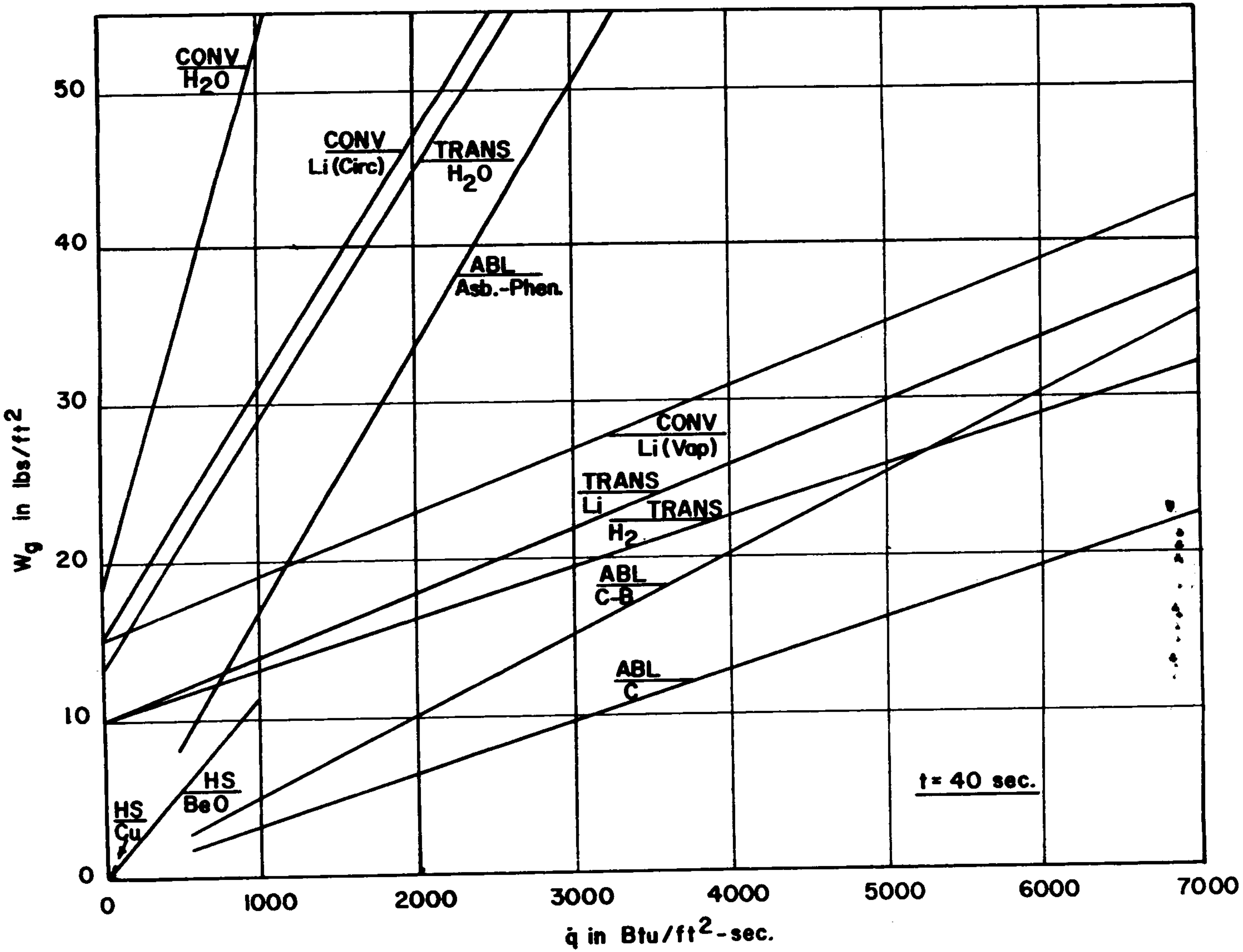

Figure 8.- Comparison of thermal protection systems on the basis of weight requirements.

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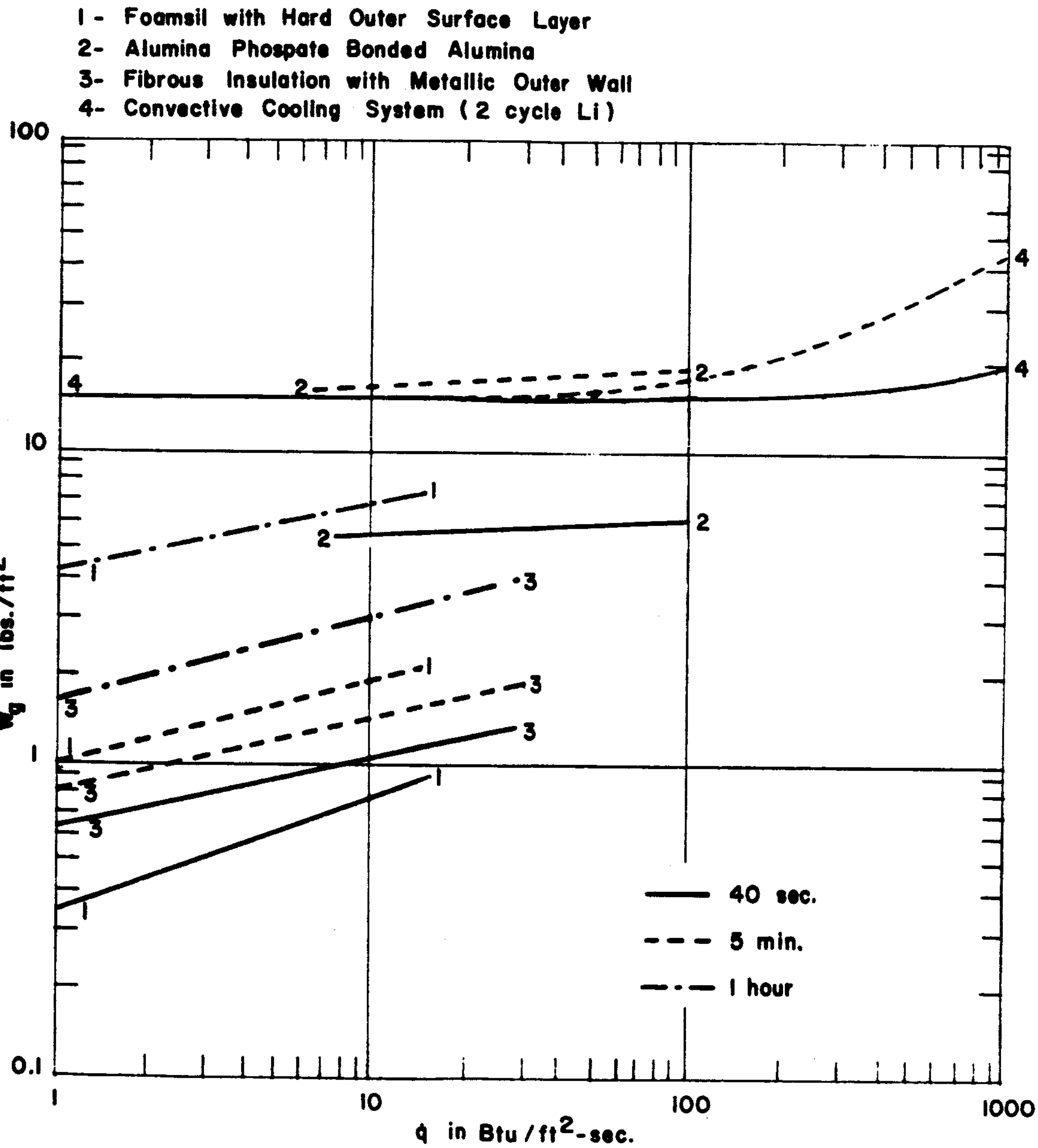

Figure 9.- Weight comparison of insulation and convective cooling (2 cycle Li) systems.

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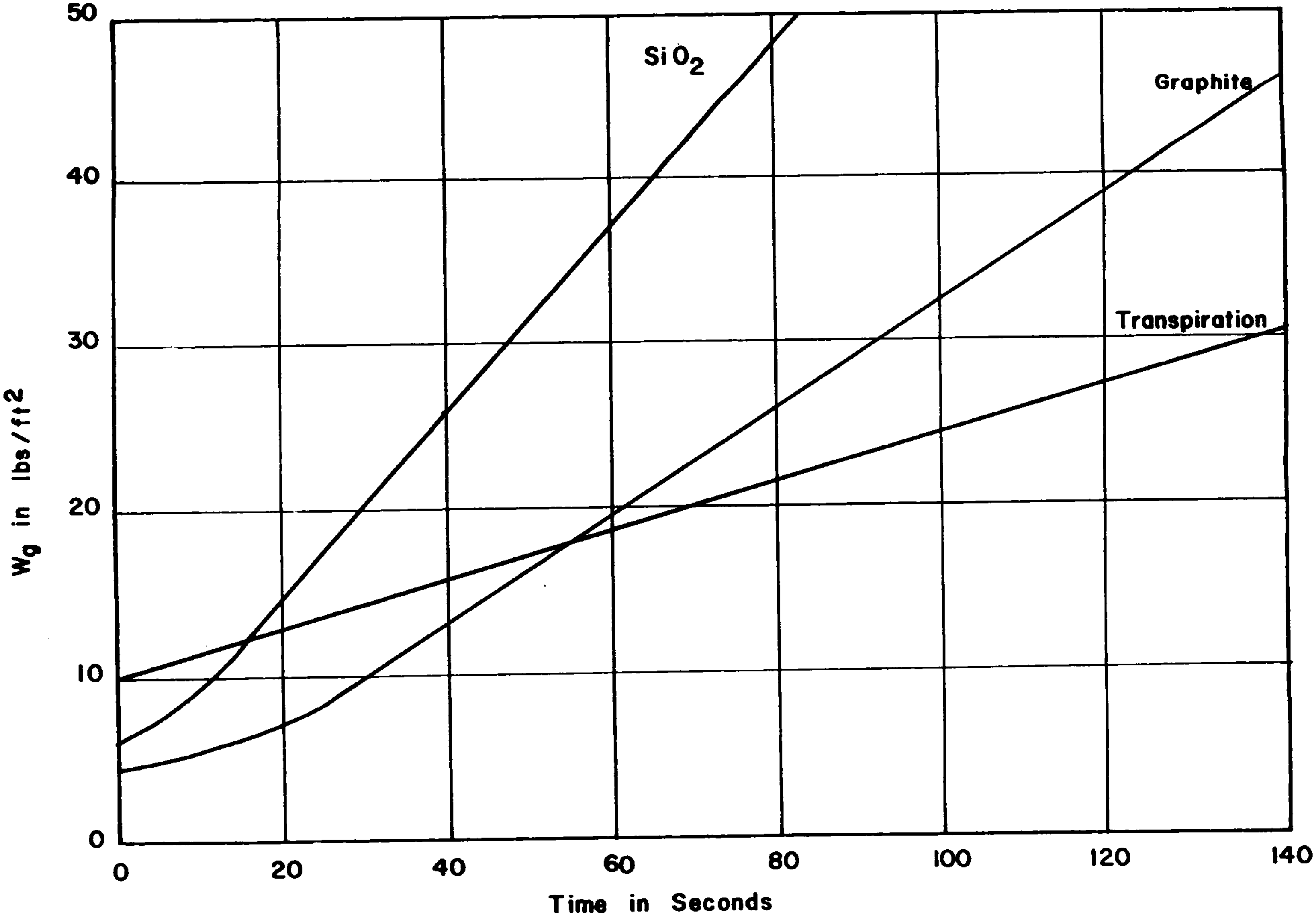

Figure 10.- Comparison of transpiration and ablation systems (from the viewpoint of operation time).

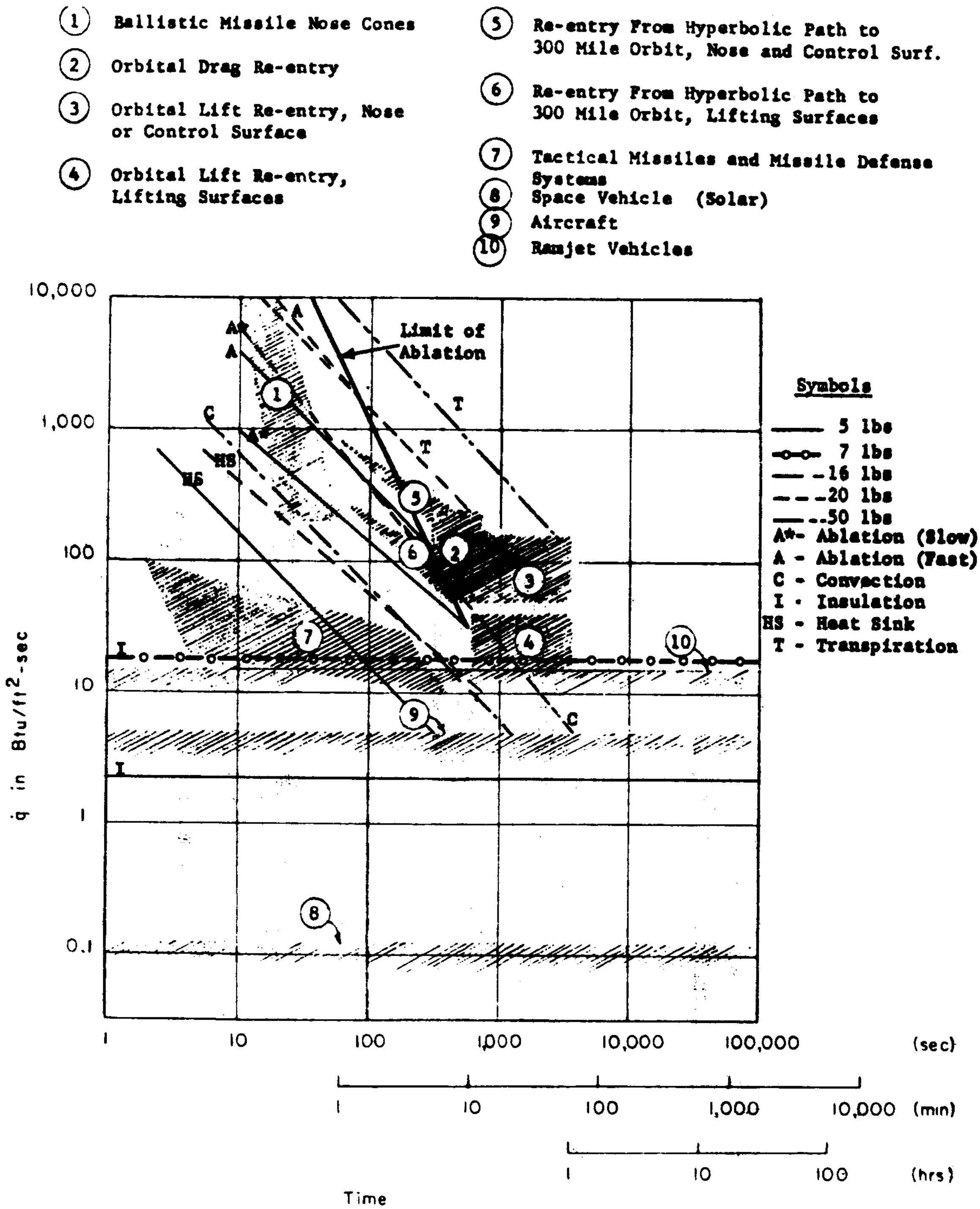

Figure 11.- Comparison of vehicle requirements and capabilities of TPS.

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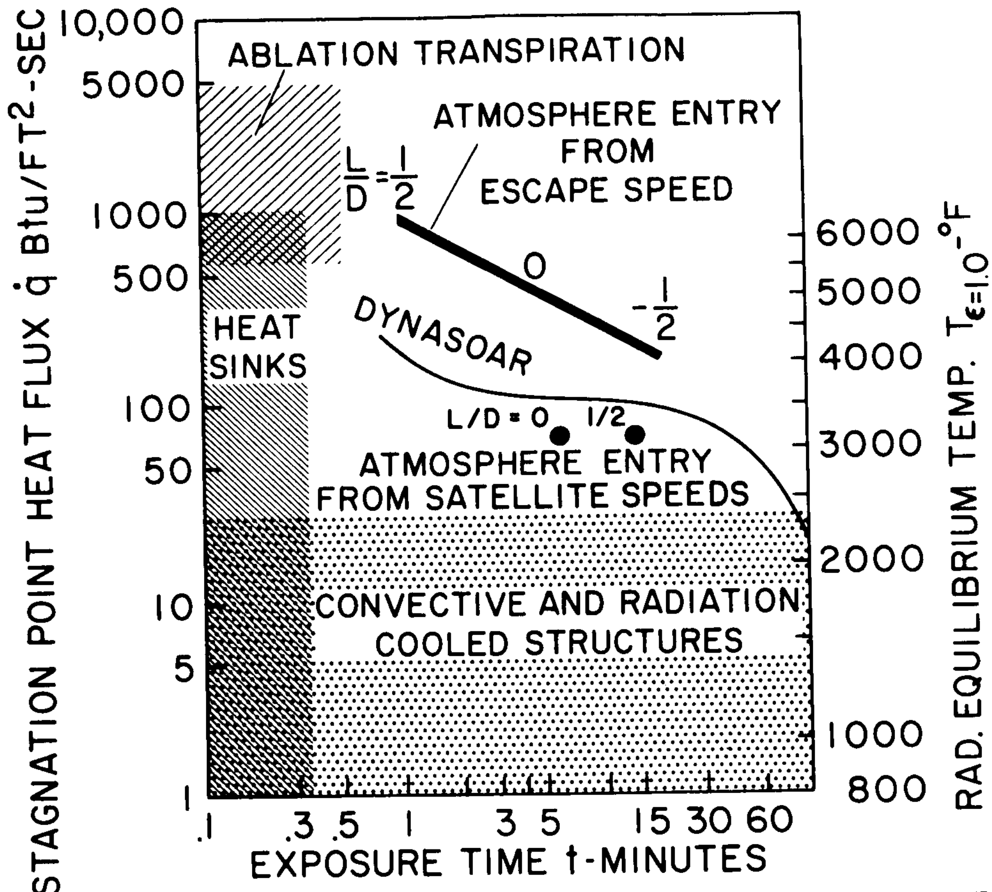

NOTE: AREAS OF APPLICABILITY OF VARIOUS TPS BY DR. W. STEURER

Figure 12.- Thermal environment for several missions.

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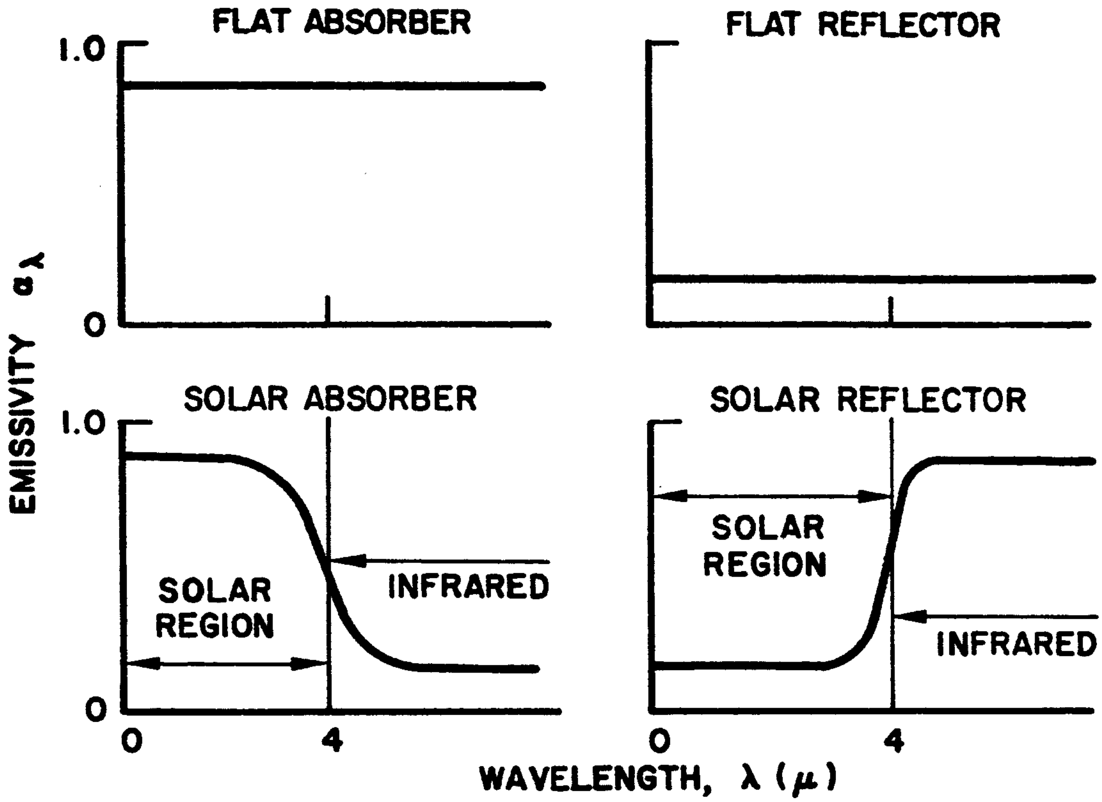

Figure 13.- Ideal stable surface finishes.

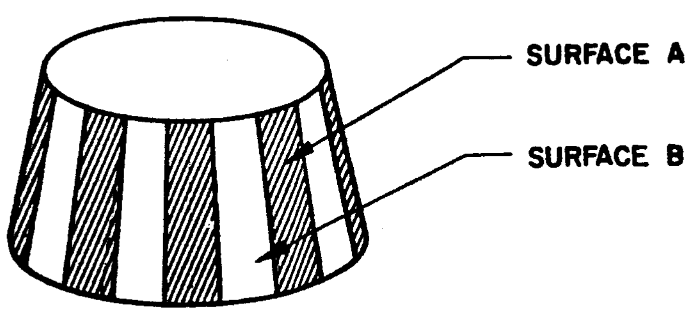

Figure 14.- Representative surface-finish mosaic.

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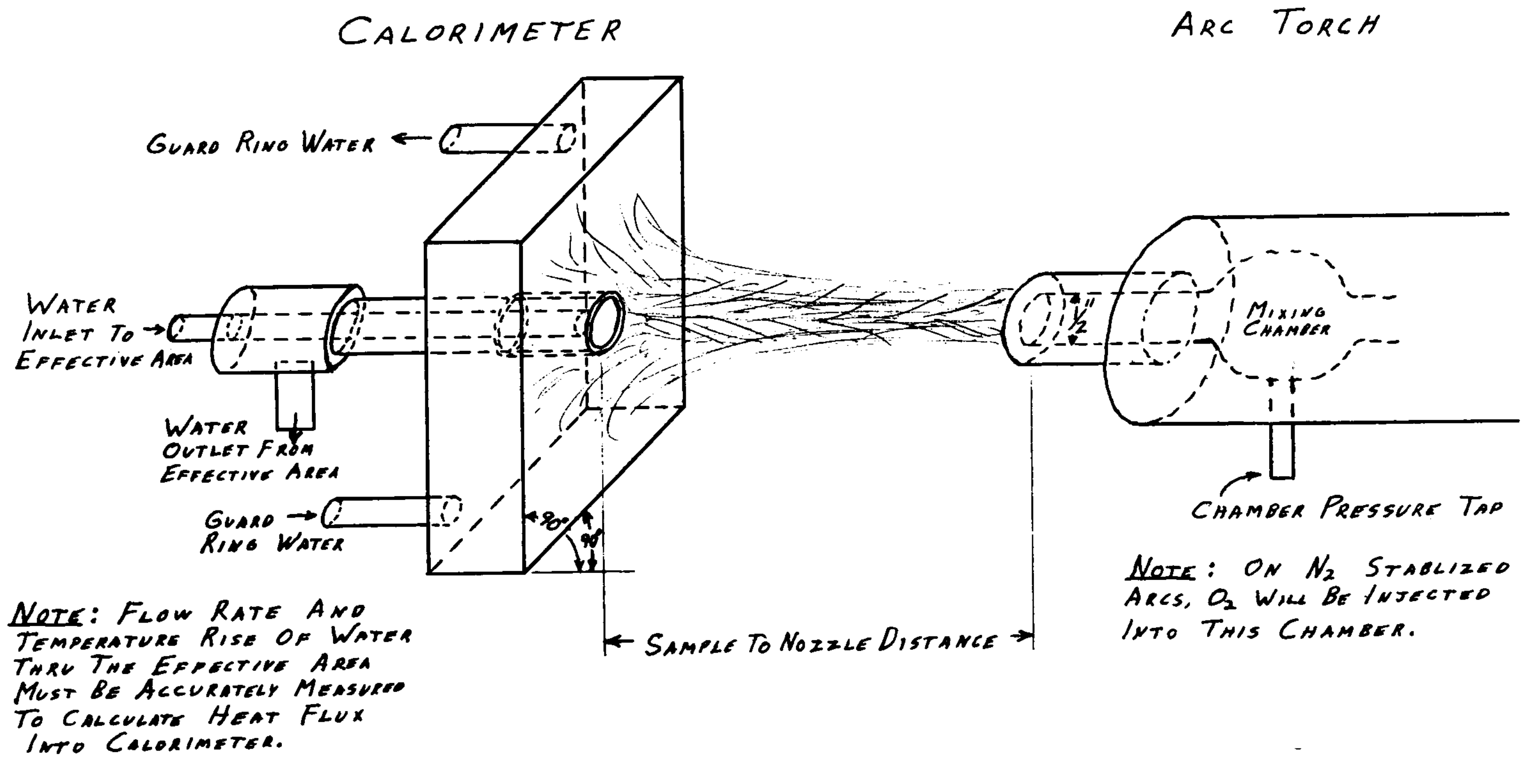

Figure 15.- Mixing chamber attachment.

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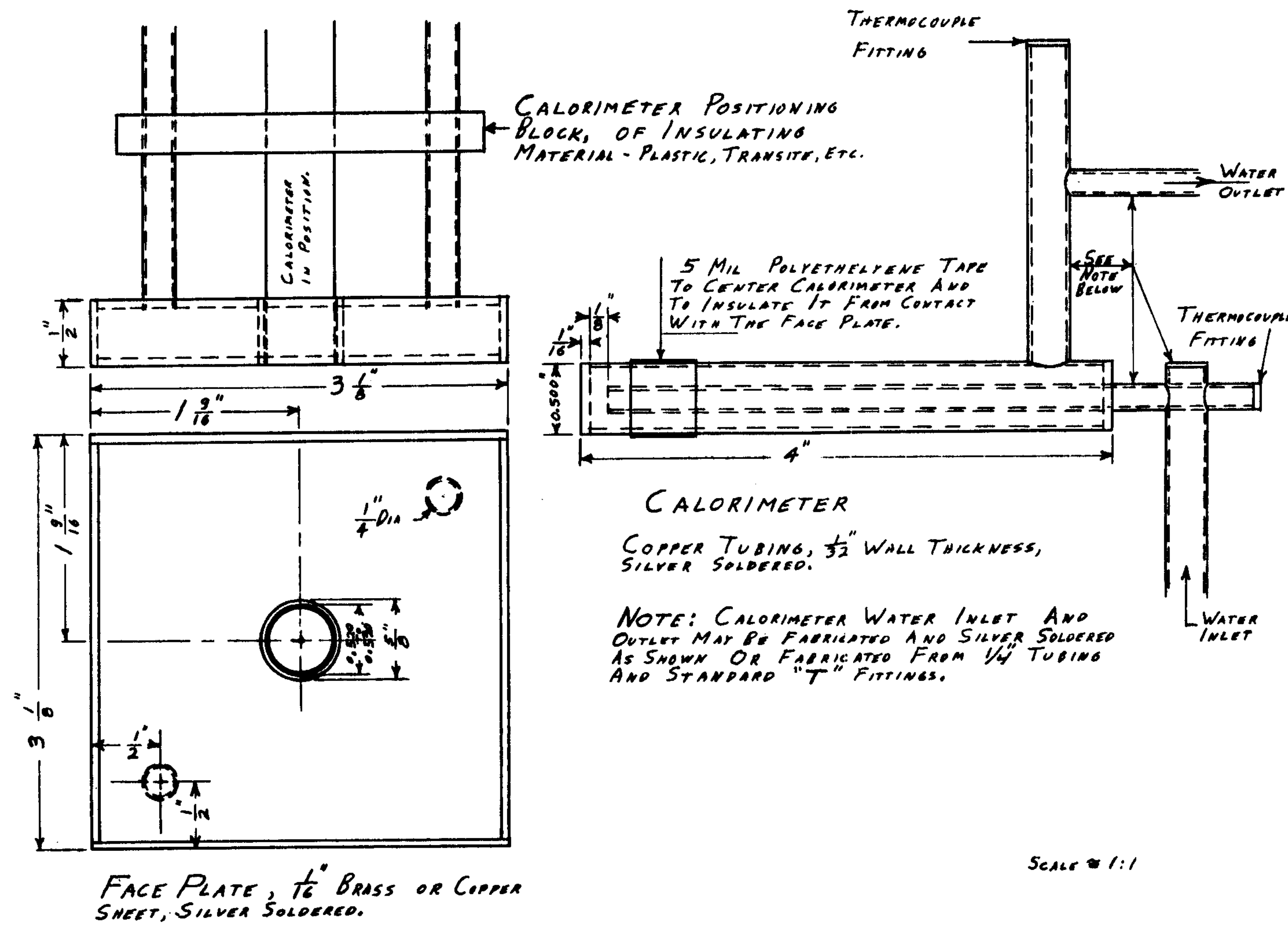

Figure 16.- Water-cooled copper calorimeter - Melpar design.

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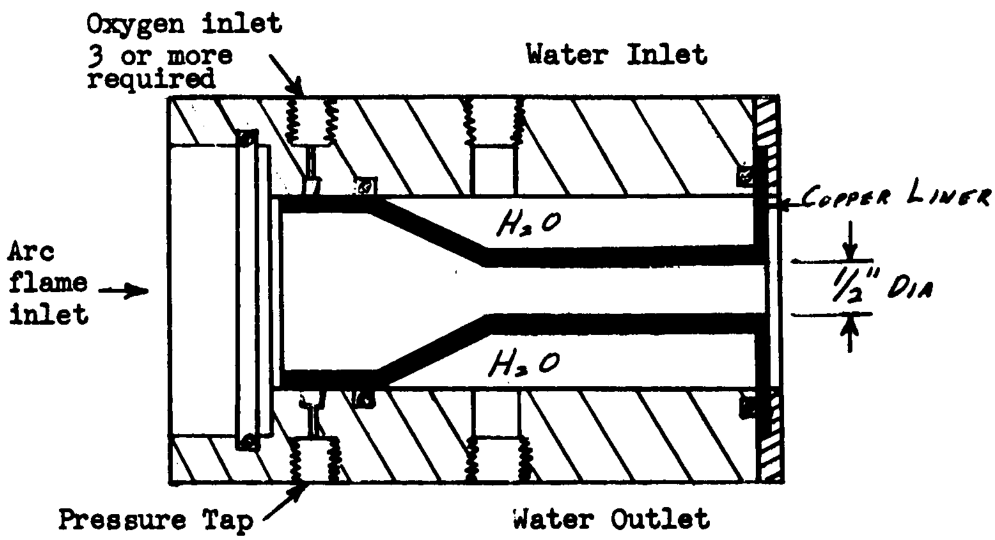

MODIFIED DELAVAL NOZZLE

Figure 17.- Subsonic nozzle and mixing chamber design; suggested by M. L. Thorpe, Thermal Dynamics Corporation. This is the same as a normal mixing nozzle except that it does not diverge.

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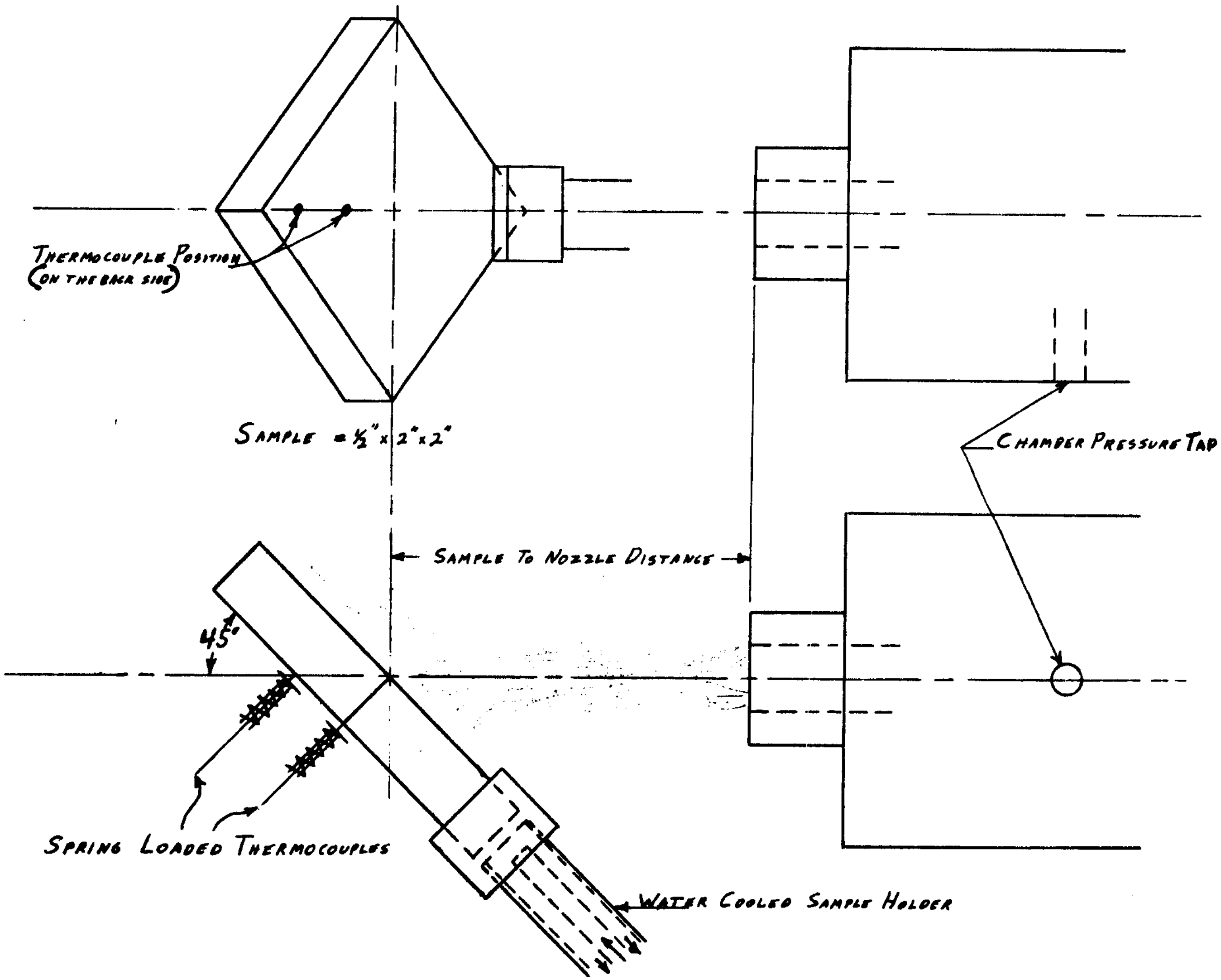

Figure 18.- Sketch of setup for testing of samples.

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Figure 19.- Data sheet - part 1.

Facility Parameters: Test no.________

Description of equipment
Electrode material, characteristics, and polarity
Nozzle description
Total power input - kw, volts, amps
Power loss to coolant - kw, Btu/sec
Power into gas - kw, Btu/sec X
Stabilizing gas composition, flow rate, and pressure - 6
lb/sec & lb/sq in 5
Mixing gas composition, flow rate, and pressure - lb/sec & lb/sq in 0
Stagnation pressure at sample - lb/sq in
Heat flux at sample - Btu/sq ft-sec
Time of exposure - sec
Nozzle chamber pressure - lb/sq in

Calculations:

Gas velocity at nozzle and at sample - ft/sec
Gas enthalpy at nozzle and at sample - Btu/lb
Gas temperature - $^{\circ}$F
Mach number

Sample parameters:

Material and dimensions - in
Weight before and after test - lb
Density before and after test - lb/cu in
Front side temperature - $^{\circ}$F. State method of measurement
and include plot of temperature versus time
Back side temperature - $^{\circ}$F
Emissivity of face
Thermal drop through sample - $^{\circ}$F
Depth of erosion - in
Volume of erosion - cu in
Distance from nozzle - in
Comments and other measurements

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Figure 19.- Data sheet - part 2.[xi,xii]

Comments on calibration: Test no.______

Arc variations with time; accuracy of measurements; calorimeter water flow rate, pressure, and temperature rise; correlation between various types of calorimeters.

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Comments on arc:

Pulsing; temperature, mass flow, velocity and pressure distributions; stability; direction and movement of flame.

Comments on test and sample:

Sample alignment, preparation, deformation, surface roughness.

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xi. Supply instantaneous E_{arc} vs I_{arc} curves showing fluctuation in power level.

xii. "Calculations": state technique usual to determine enthalpy; determine enthalpy by at least two methods out of those listed below:

1. Equilibrium sonic flow
2. First law
3. Heat-transfer correlation
4. Free-stream pressure and velocity

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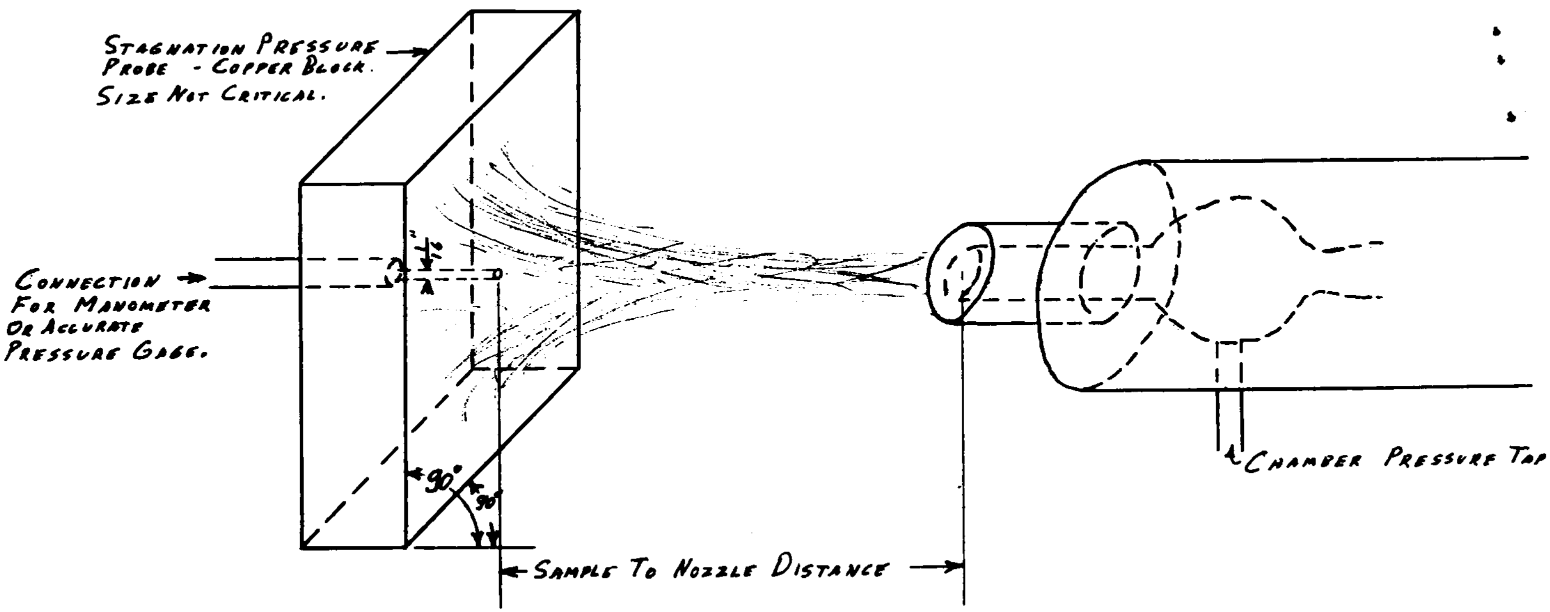

Figure 20.- Sketch of setup for stagnation-point pressure measurements.

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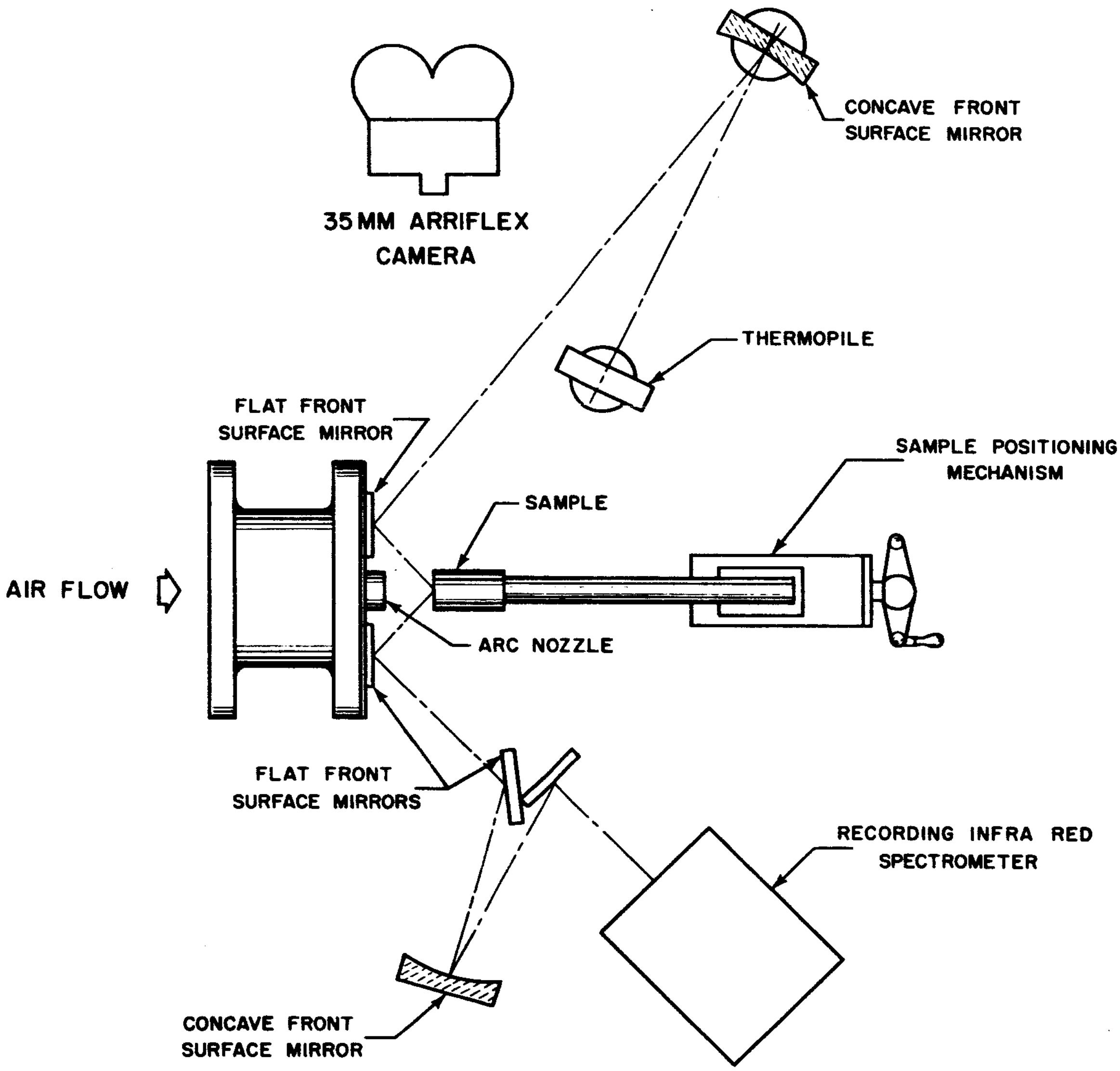

Figure 21.- Experimental apparatus for the determination of the thermochemical heat of ablation.

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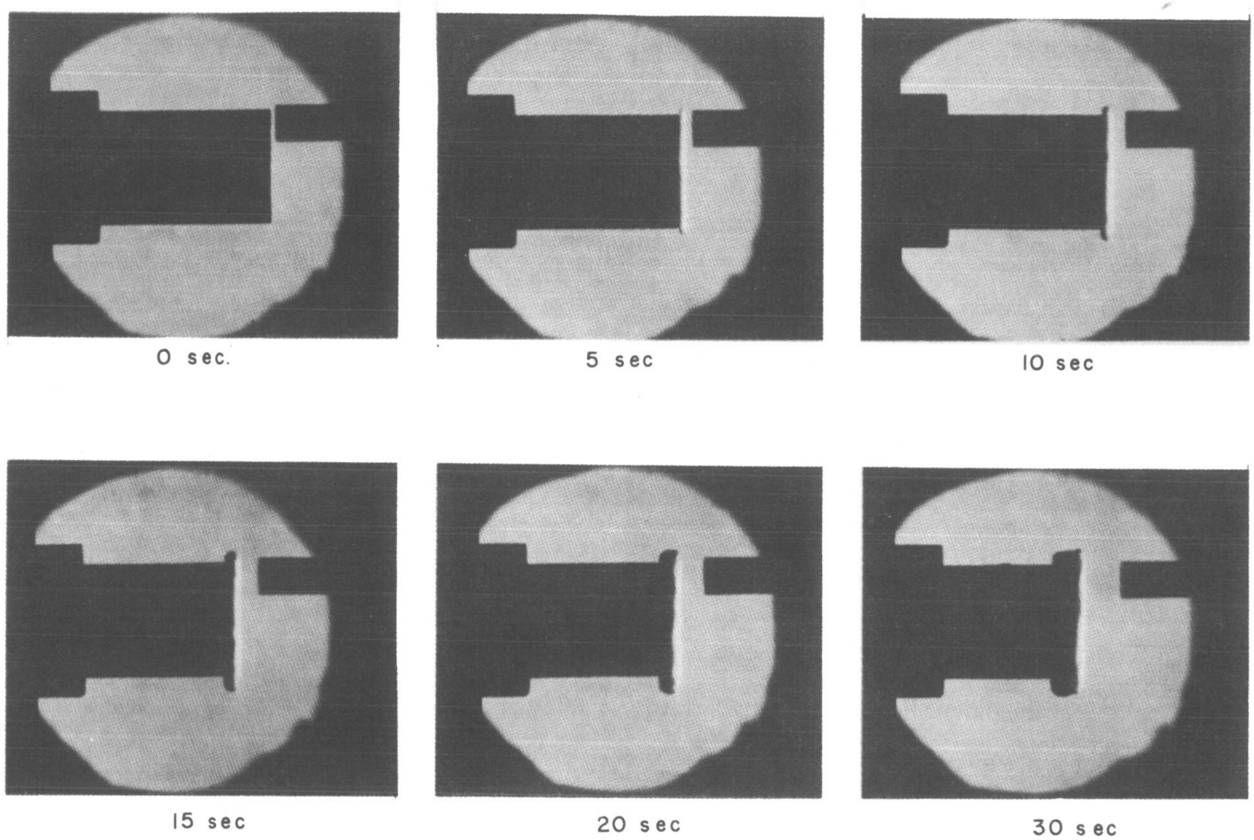

Figure 22.- Film sequence - silica (modified). L-61-8411

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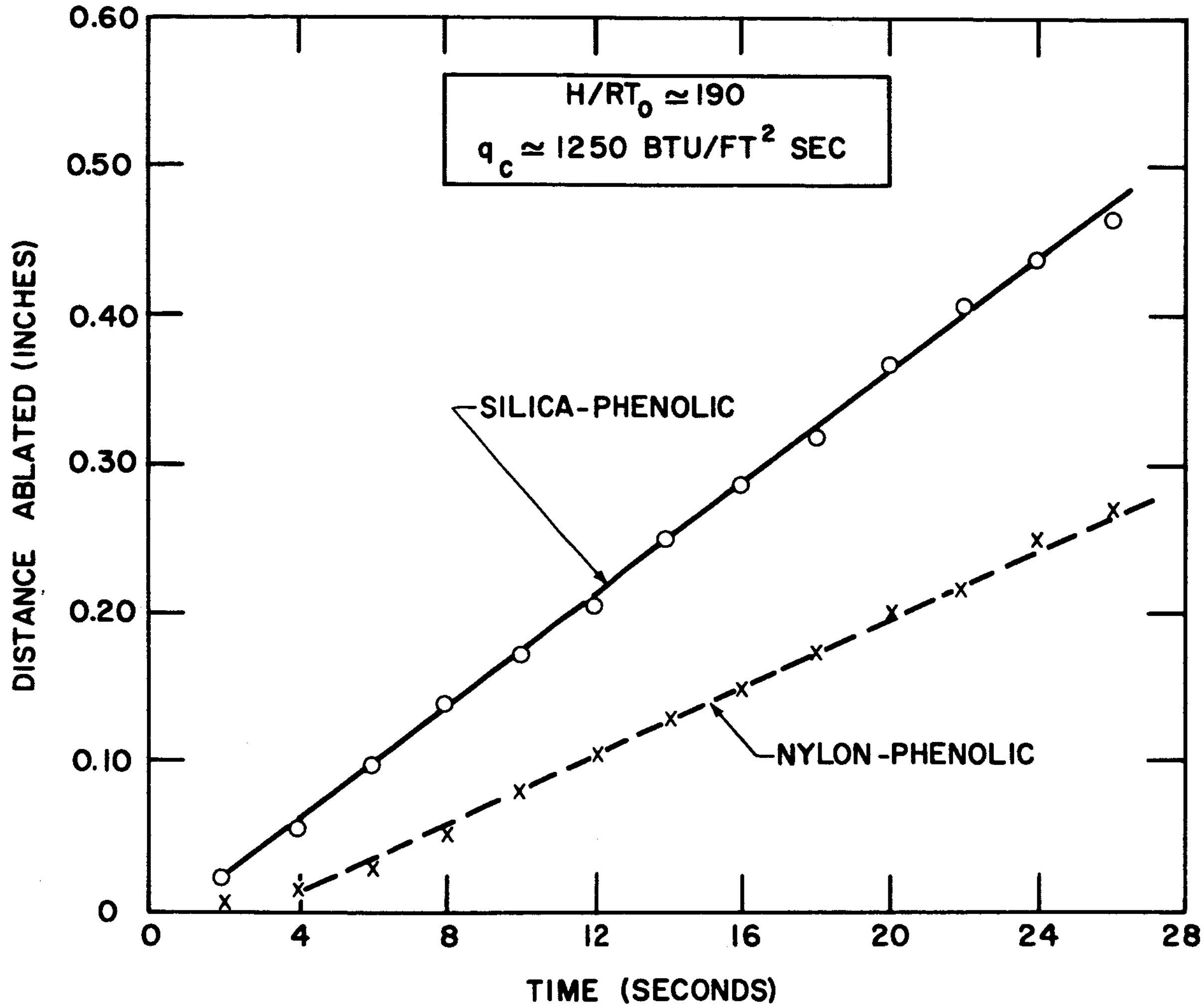

Figure 23.- Distance-time history of sample surface.

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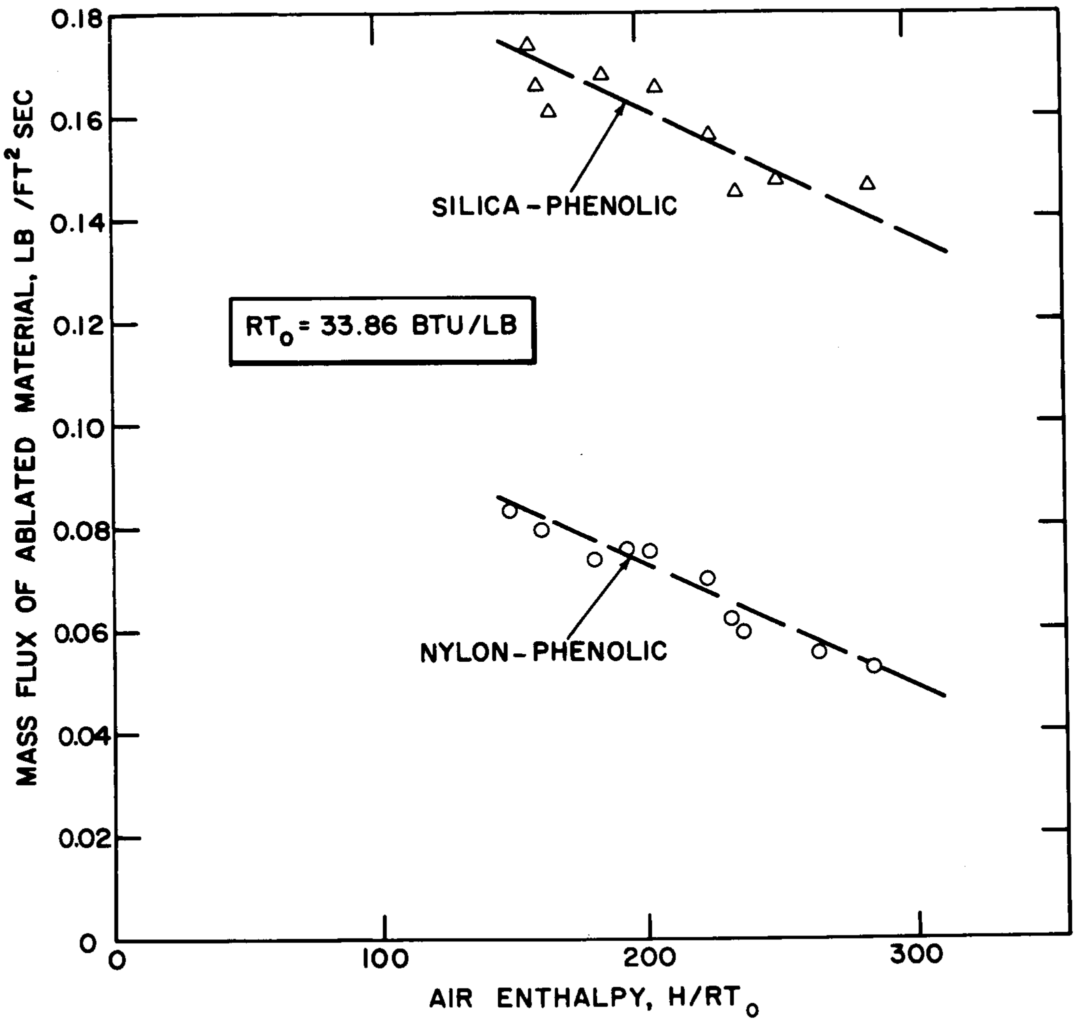

Figure 24.- Mass flux of ablated material versus air enthalpy.

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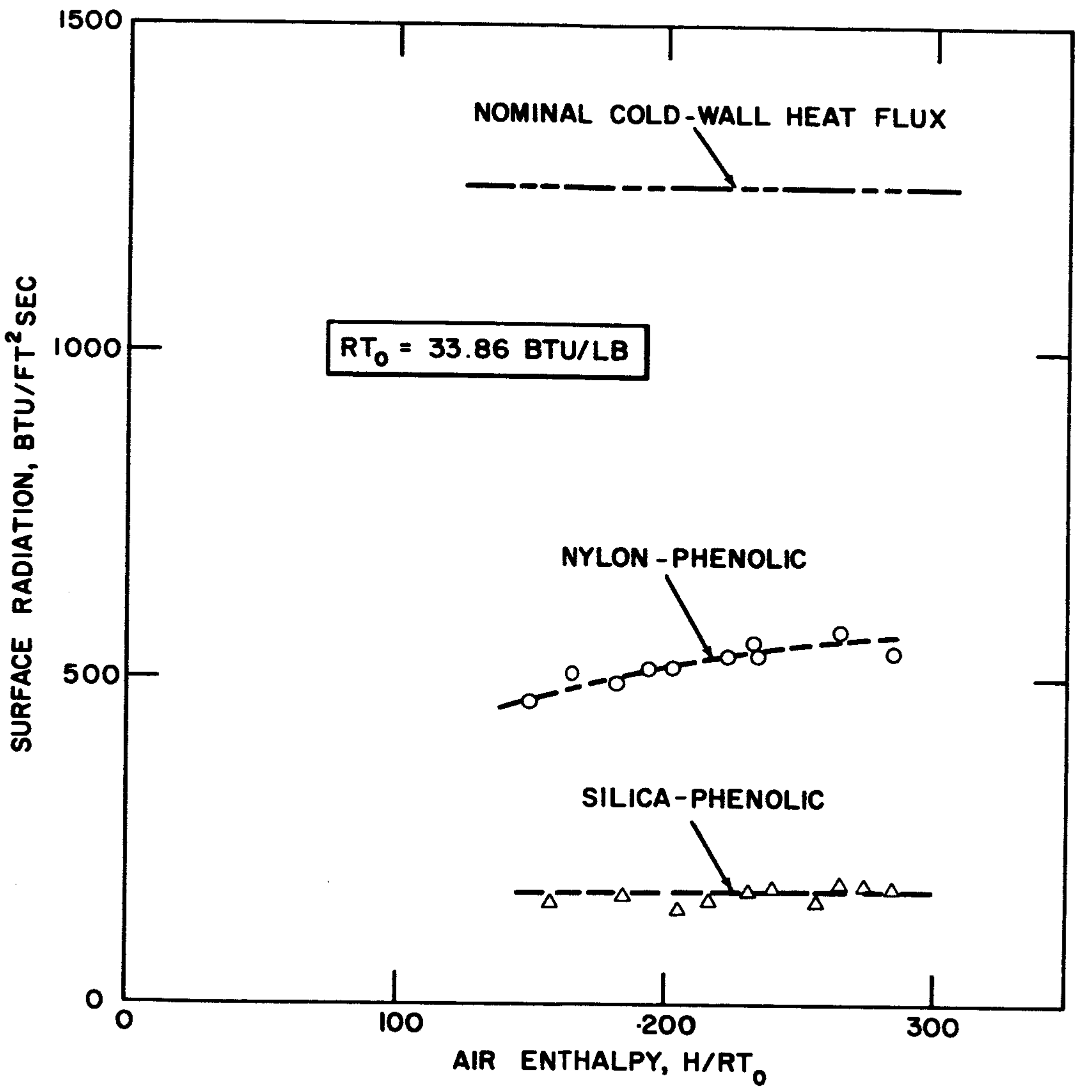

Figure 25.- Surface radiation versus air enthalpy.

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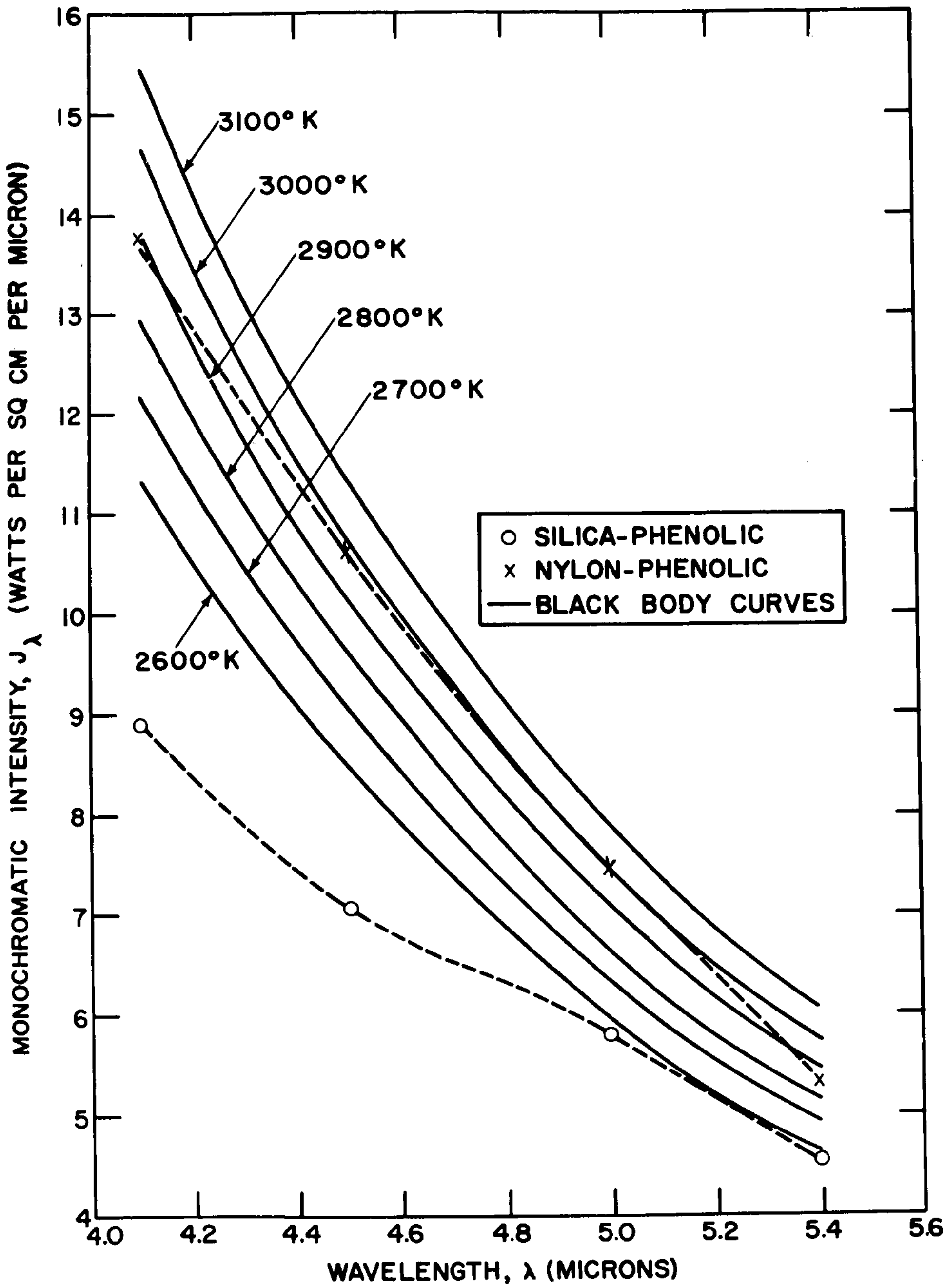

Figure 26.- Spectral radiation distribution.

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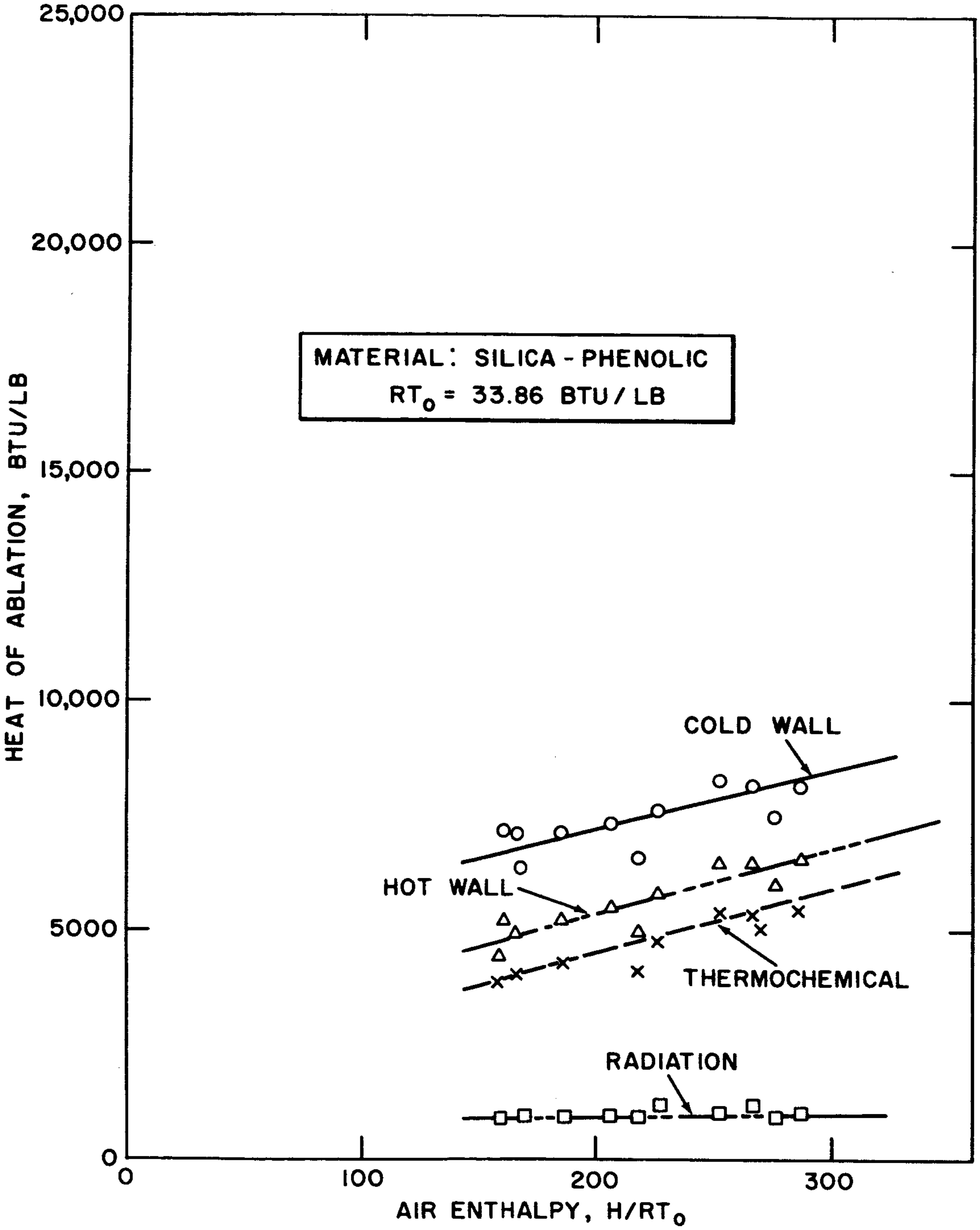

Figure 27.- Heat of ablation versus air enthalpy; silica-phenolic.

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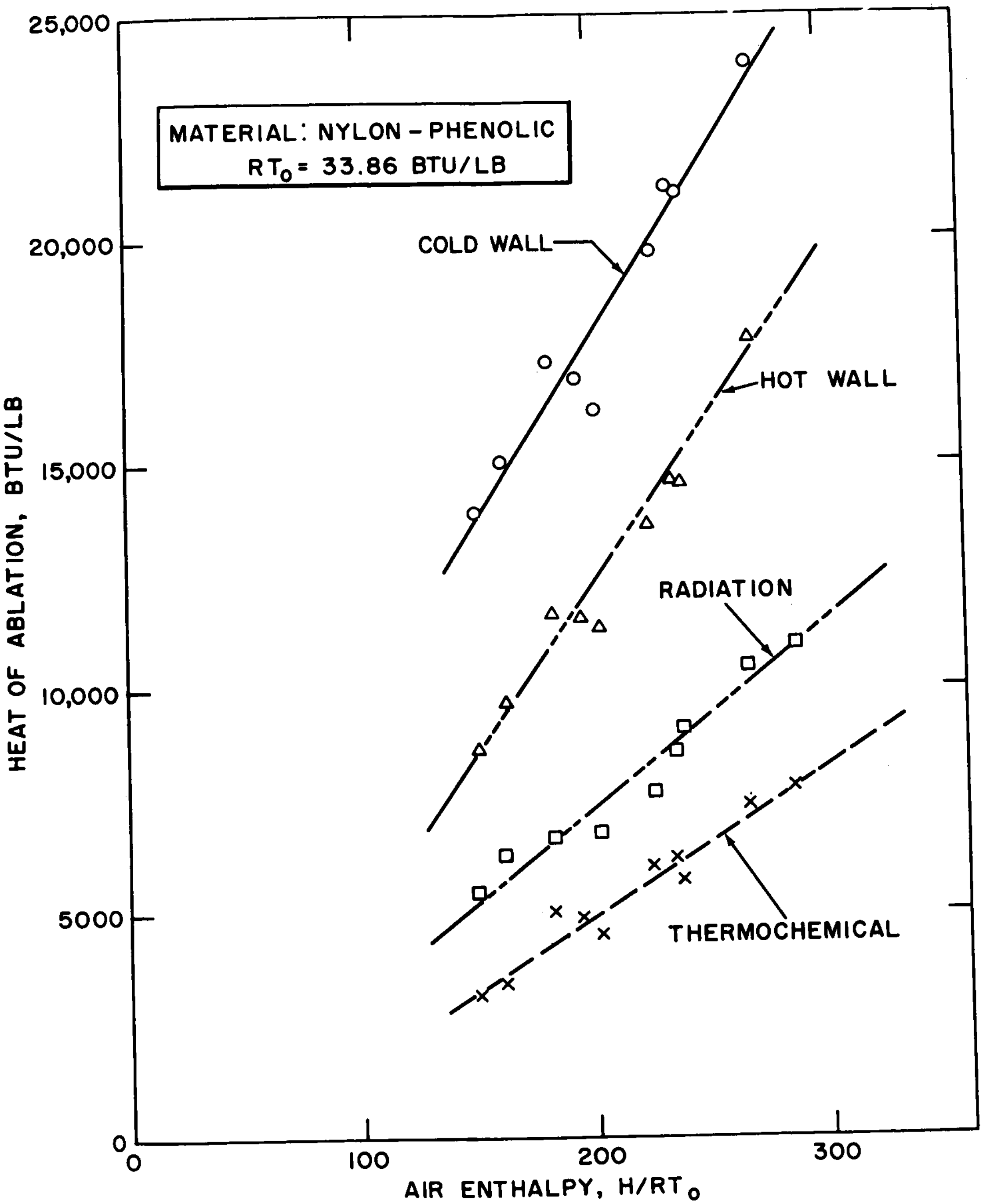

Figure 28.- Heat of ablation versus air enthalpy; nylon-phenolic.

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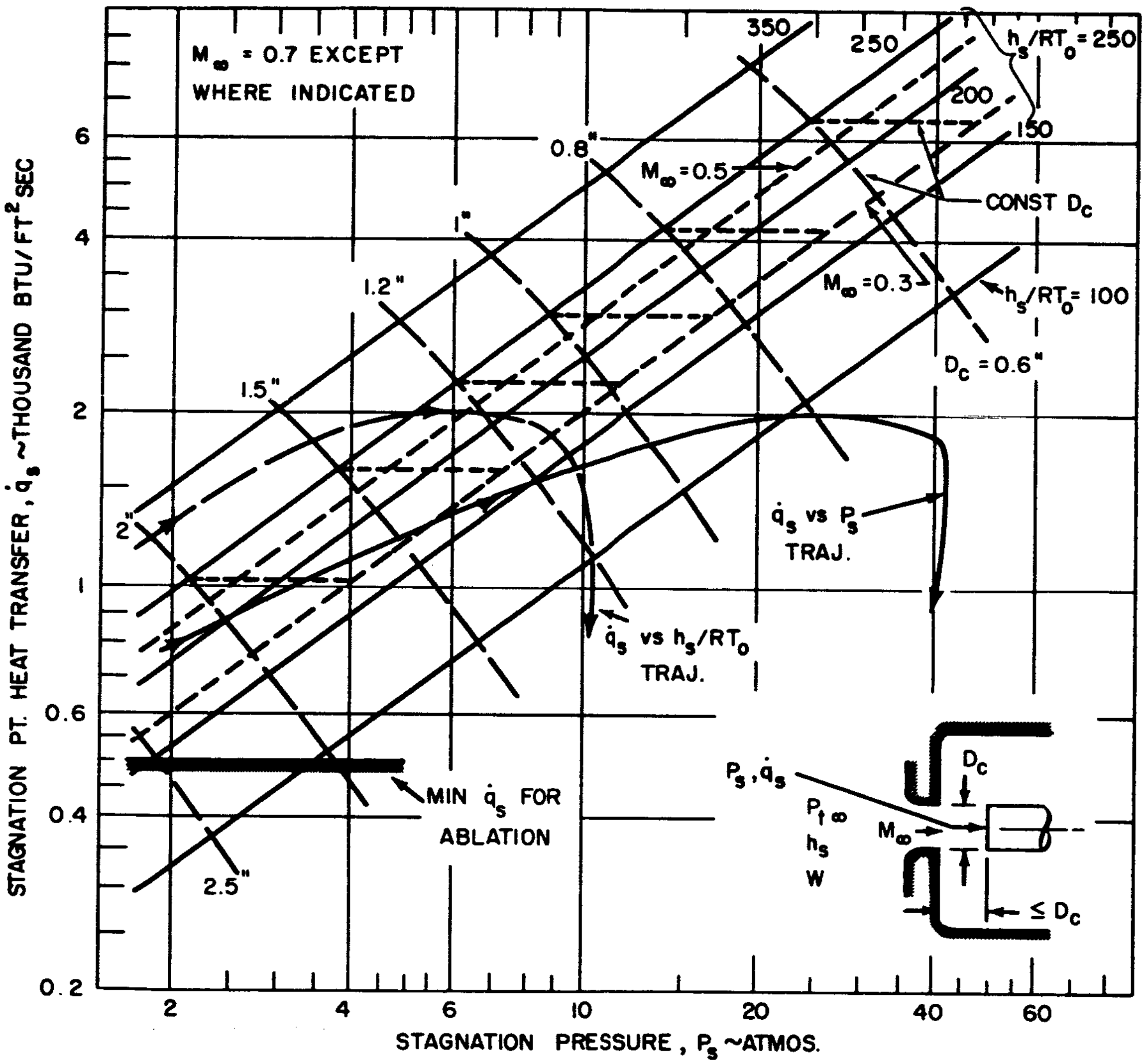

Figure 29.- Splash capabilities, power-to-air = 3 megawatts.

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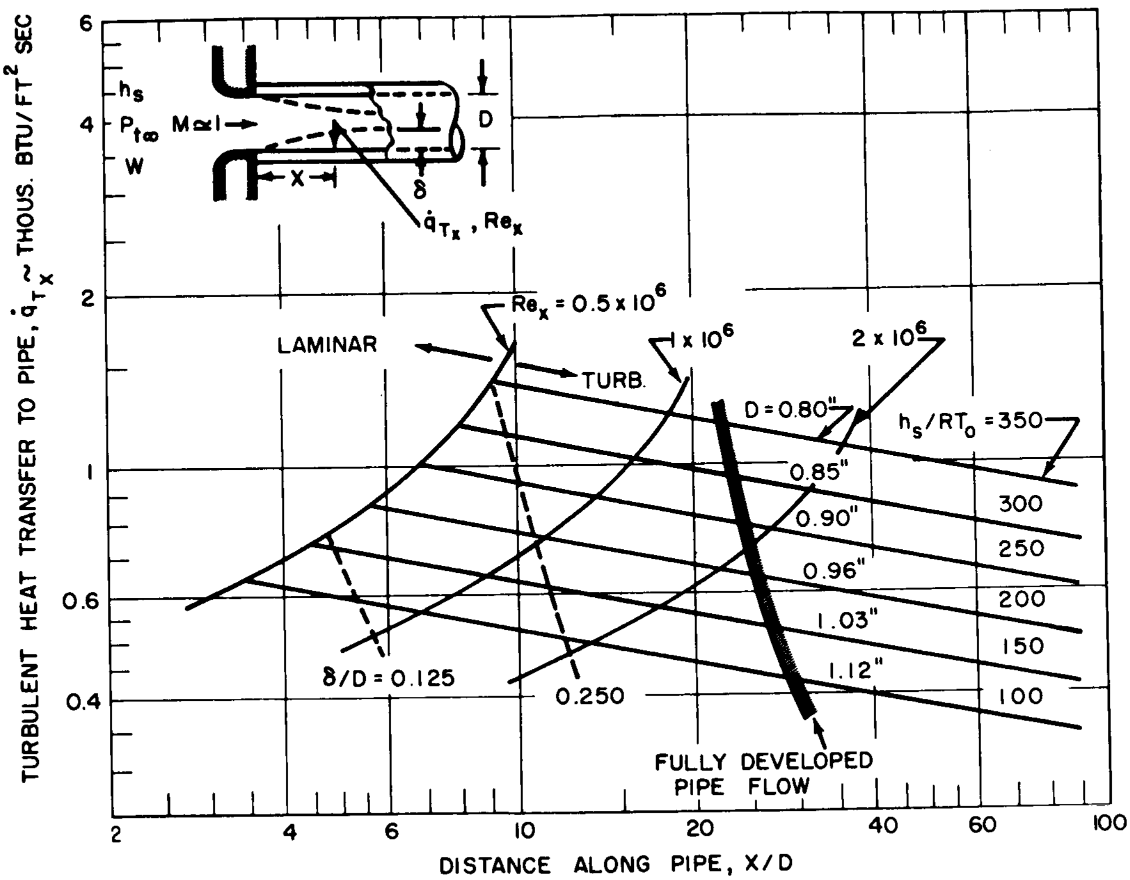

Figure 30.- Turbulent pipe capabilities, power-to-air = 3 megawatts; $P_{t\infty}$ = 10 atmospheres.

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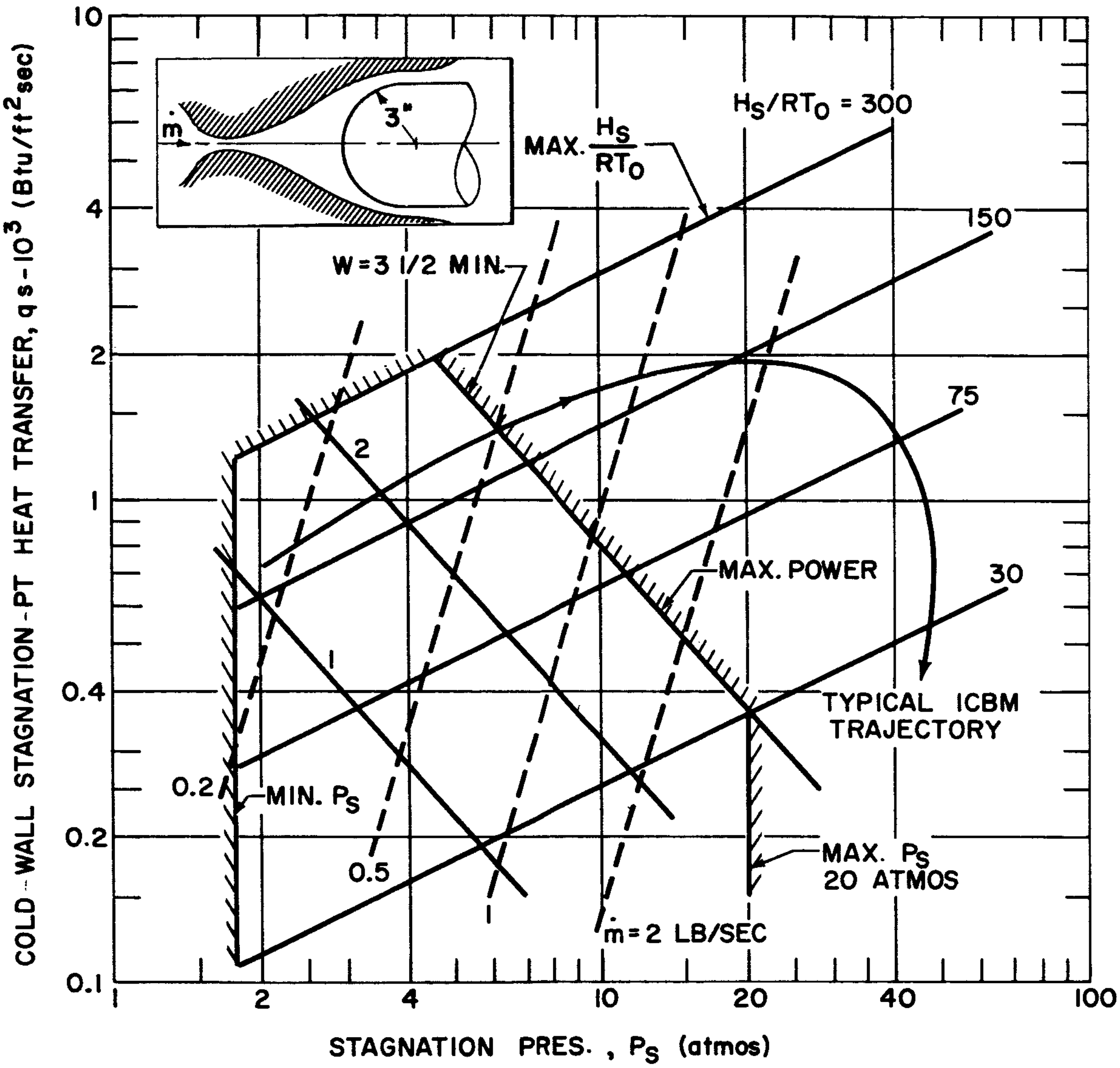

Figure 31.- Ten-megawatt arc simulation with 6-inch-diameter shroud model.

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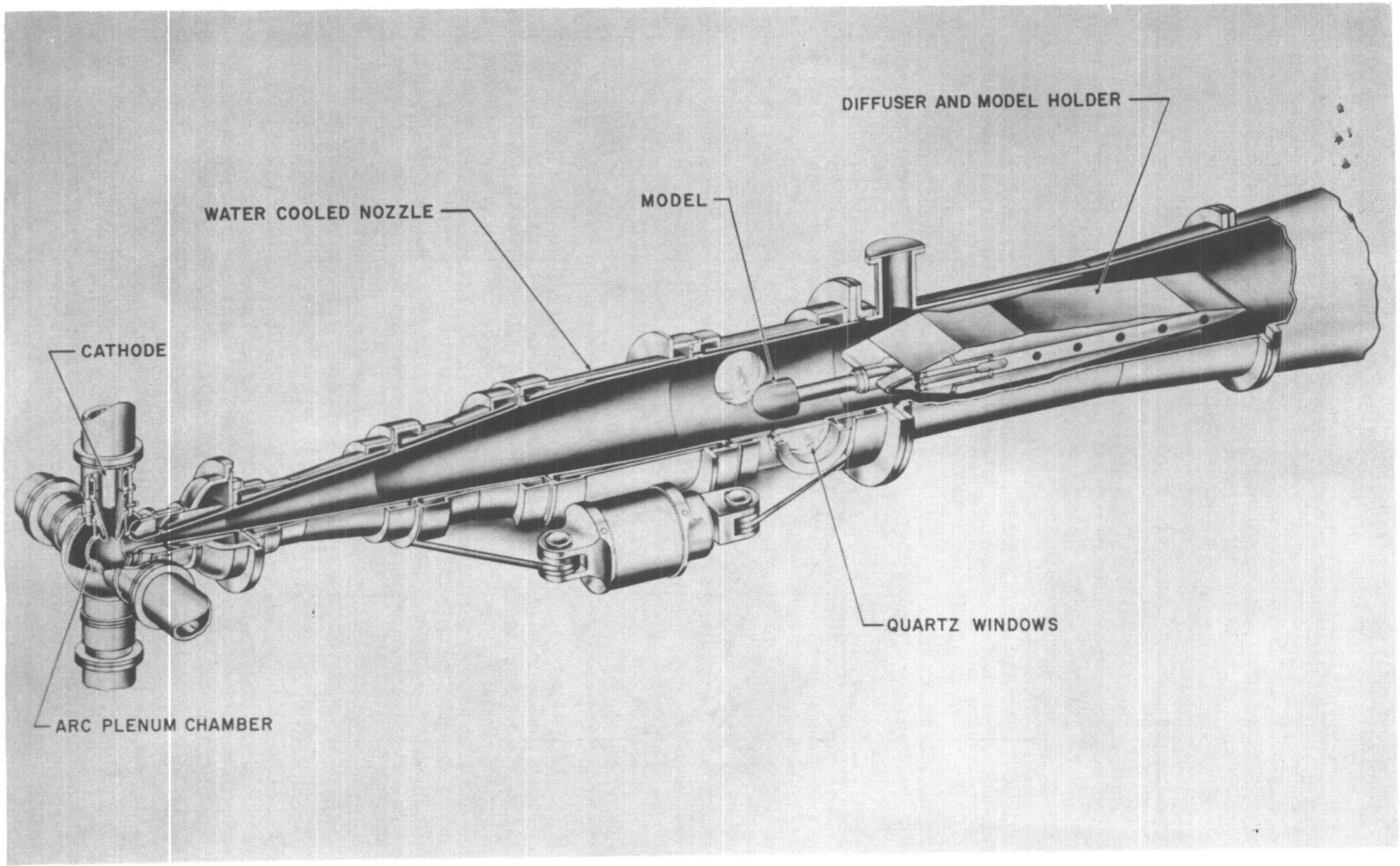

Figure 32.- Cutaway of hyperthermal wind tunnel. L-61-8412

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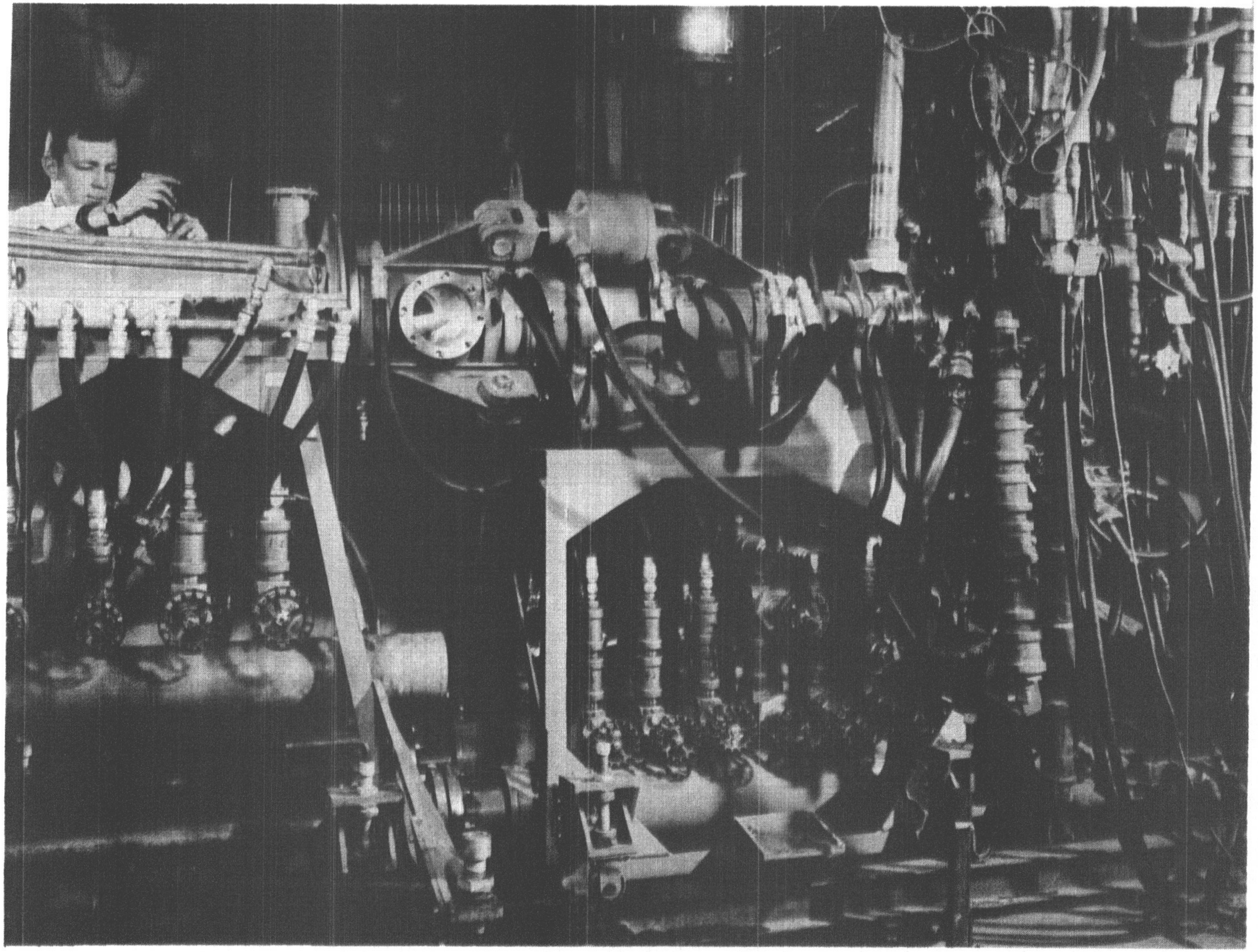

Figure 33.- Hyperthermal wind tunnel being connected to 10-mgw arc. L-61-8413

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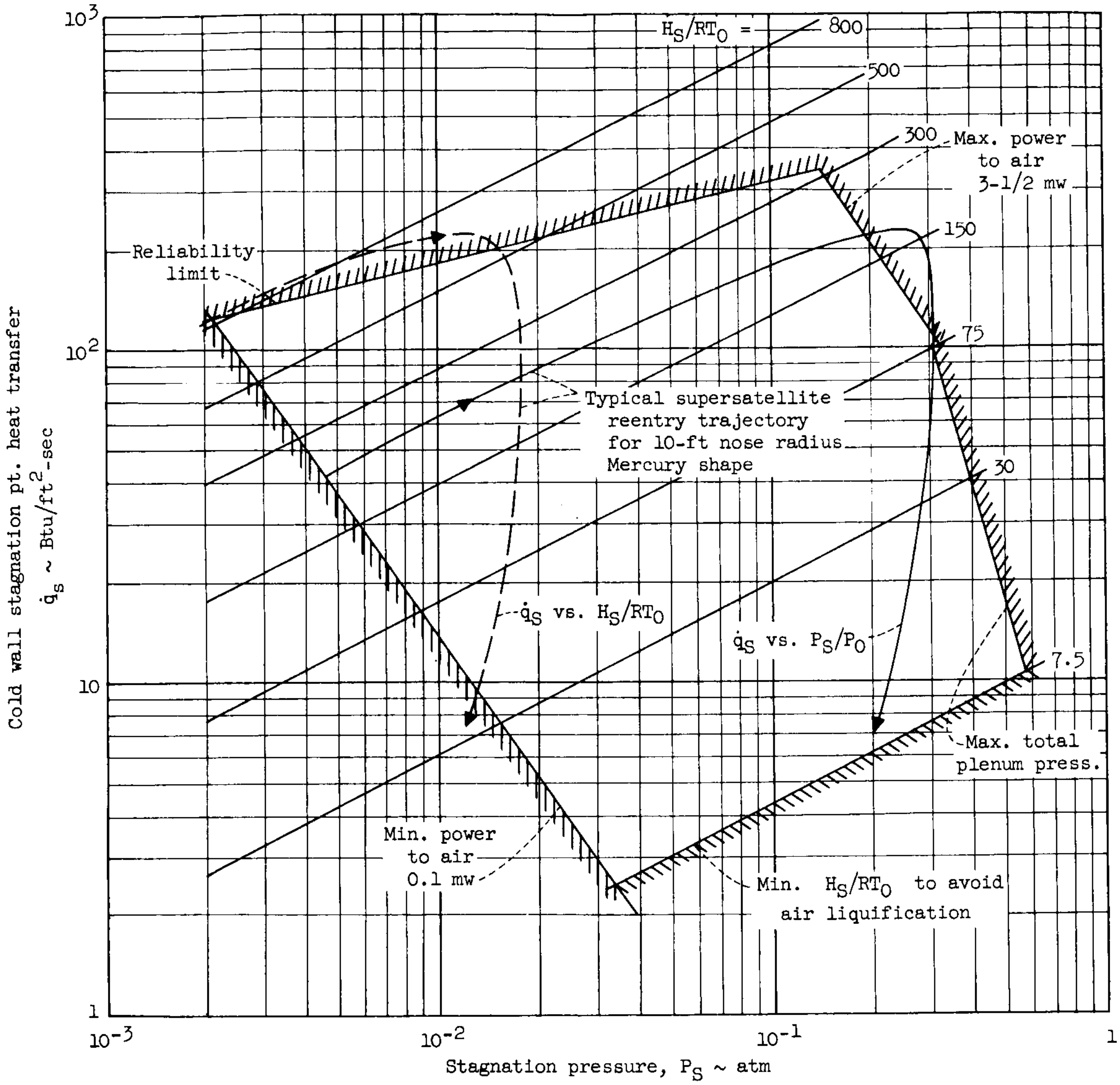

Figure 34.- Simulation capability of hyperthermal wind tunnel.

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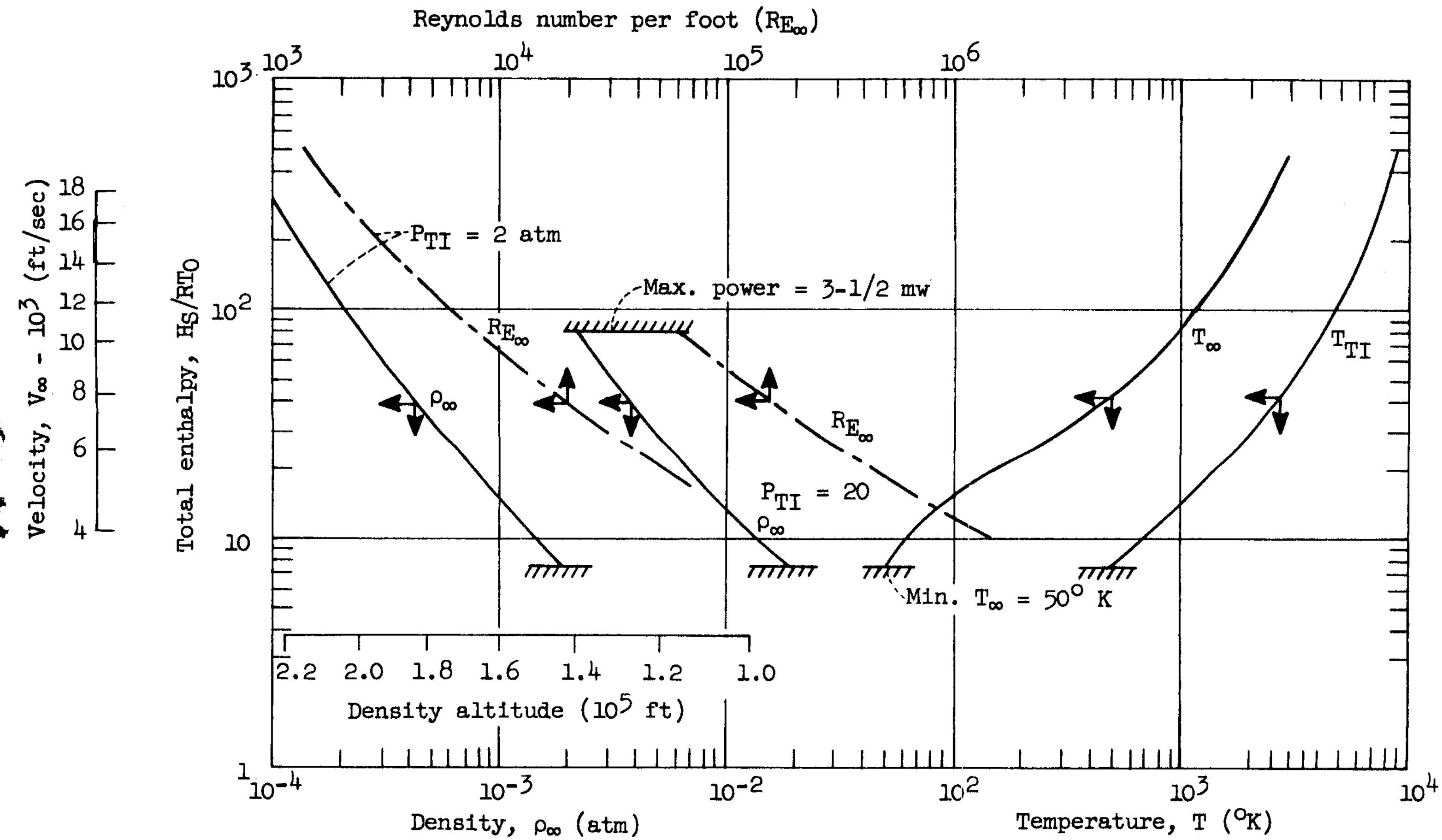

Figure 35.- Hyperthermal wind-tunnel test-section conditions.

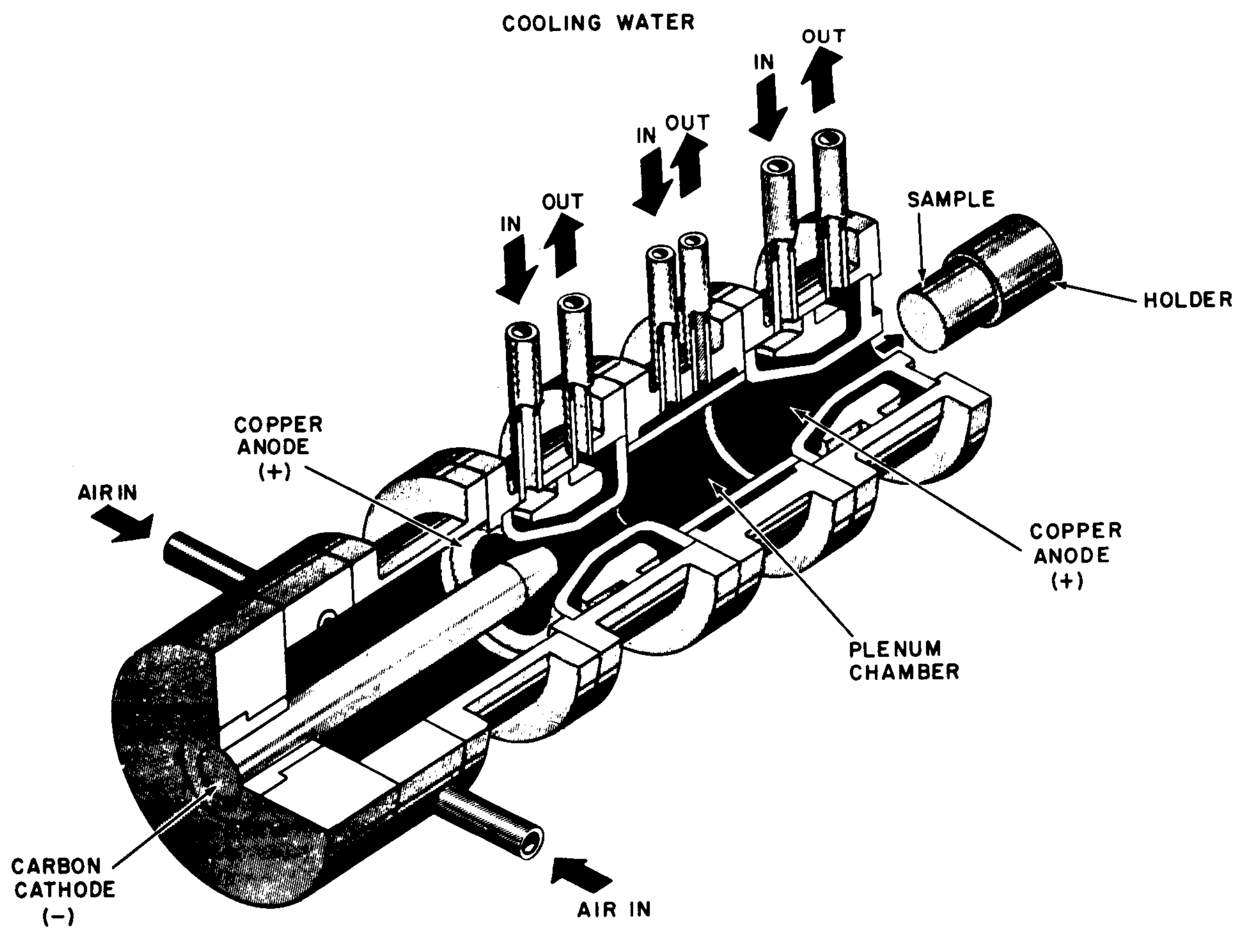

Figure 36.- Schematic of arc-jet unit.

REFERENCES

Many of the excellent technical reports on thermal protection systems which have been issued by industrial and government organizations have not been included. As an aid in keeping up with ablation literature, the Nonmetallic Materials Laboratory, ASD, will soon have a complete list of applicable references which can be obtained by qualified requesters.

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